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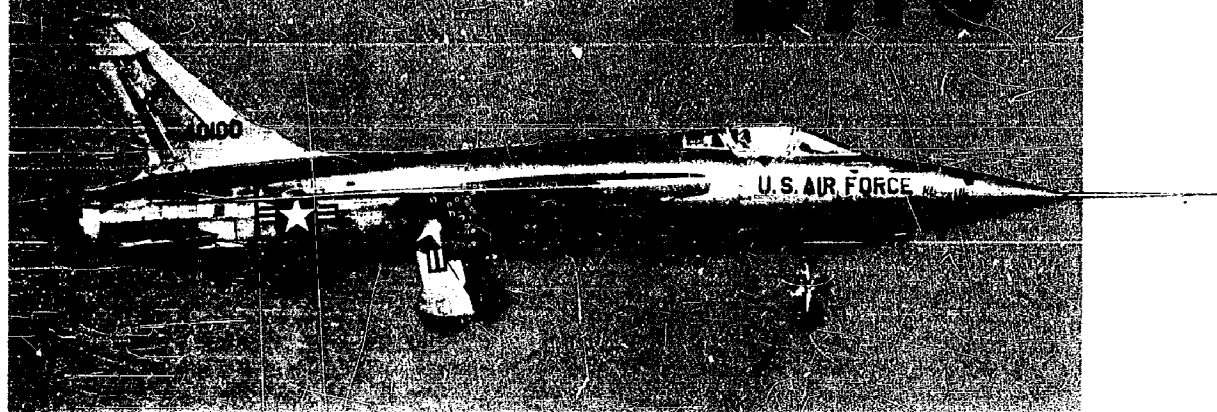
PHASE II

FLIGHT EVALUATION

PHASE I FLIGHT
EVALUATION SUMMARY

PHASE II FLIGHT
EVALUATION SUMMARY

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F-105B



FLIGHT

AFFTC-TR-57-9 June 1957

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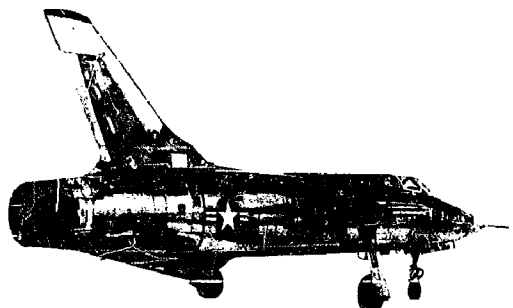
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ABSTRACT

The performance of the F-105B is superior to any fighter-bomber presently in service; however, the aircraft has a large number of deficiencies. Although a maximum speed of 1.95 Mach number can be attained, poor acceleration characteristics provide little tactical utility from speeds above 1.8 Mach number. The specific range of the aircraft is approximately 20 percent less at 35,000 feet than estimated by the contractor. Both the longitudinal and lateral control systems are unsatisfactory. Structural failures which occurred during the test program, such as the boundary layer splitter and speed brake skin failures, must be investigated and action taken to prevent future occurrence. A major redesign will be required to correct cockpit deficiencies. Fixes for these and other unsatisfactory areas must be developed and incorporated before the aircraft can be considered acceptable for tactical use.

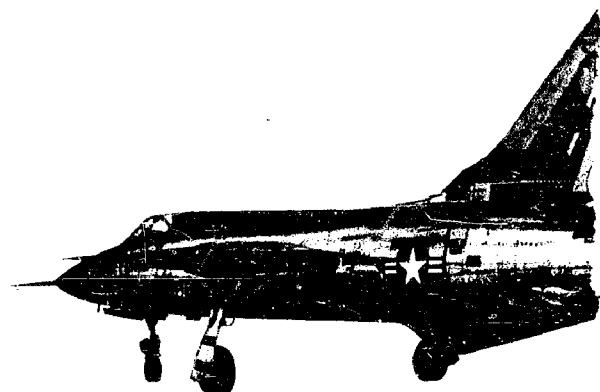
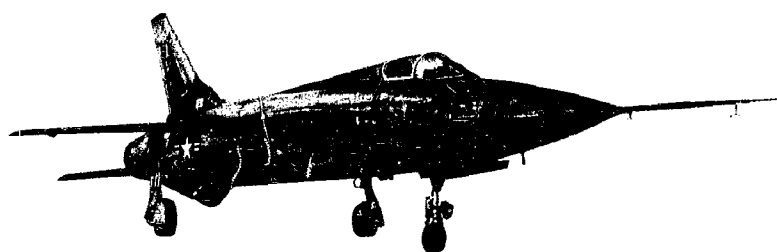
*This report
has been
reviewed and
approved,*
23 MAY 1957

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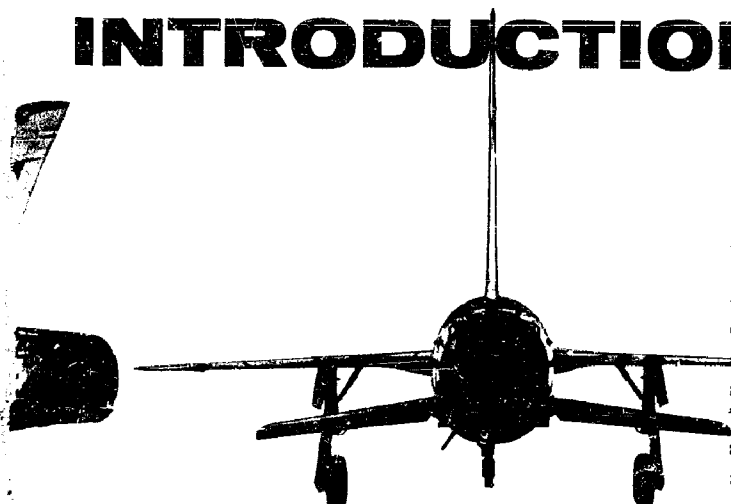
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INTRODUCTION



This report presents the results of the Phase II flight test of the F-105B, USAF S/N 54-100. The program was performed under the authority of Commander, Air Force Flight Test Center as requested by Headquarters, Air Research and Development Command, to obtain preliminary performance, stability and control data and qualitative information on the handling characteristics of the airplane.

The test program was conducted at the Air Force Flight Test Center, Edwards Air Force Base, California between 8 January and 7 March 1957. Eighteen flights (13:45 hours) were made to obtain quantitative data. The last two weeks were utilized to obtain nine flights for qualitative evaluation by pilots of the Air Proving Ground Command, Tactical Air Command, and Air Force Flight Test Center. Two additional flights were made using F-105B S/N 54-101. One 40 minute flight was made on 28 March to evaluate the LABS and dive bombing capabilities of the aircraft, and one flight was made on 24 April to evaluate night operation.

The F-105B is a single place, swept wing and empennage fighter-bomber with a fuselage conforming to the area rule. The aircraft is built by Republic Aviation Corporation and is powered by one Pratt and Whitney J75-P-3 axial flow gas turbine engine equipped with an afterburner. Rated sea level static thrust of the engine is 15,500 pounds without afterburning and 23,500 pounds with afterburner. The engine has a multi-stage, two unit compressor and a three stage turbine. A compressor air bleed system is utilized to direct part of the low pressure compressor air overboard at low engine rpm to facilitate starting and provide fast engine acceleration. A control system incorporated in the intake ducts provides maximum efficiency during operation at supersonic speeds by varying the duct throat area, bleeding air overboard to match the inlet air flow to engine requirements. Control of this system is automatic, but manual control is available to the pilot. Leading edge and trailing edge flaps are provided as high lift devices. A four section speed brake on the aft end of the fuselage and a landing drag chute are provided as drag increasing devices. The speed brakes form the convergent-divergent engine exhaust nozzle. The cockpit has a clamshell canopy with an upward ejection seat system. Stores are carried in an inclosed bomb bay in the fuselage and on external wing and fuselage pylons. A 350 gallon fuel tank mounted in the bomb bay was used in the Phase II program.

Control is provided by an all-moving stabilizer, a flap type rudder, and a combination of ailerons and spoilers for lateral control. The controls are irre-

versible, and control feel is provided by spring-loaded artificial feel devices which vary the load felt by the pilot in proportion to the deflection of the stick or rudder pedals. Pitch and yaw dampers are provided for stability augmentation.

Lateral trim is provided by an electric actuator which moves a trim tab located on the left wing trailing edge flap, and an electrically-operated spoiler trim device which repositions the stick booster control valve and spring capsule. Both trim devices are controlled by a trim switch on the control stick. Trimming of the rudder and horizontal stabilizer is accomplished by electric actuators which position the respective artificial feel device to a new neutral position.

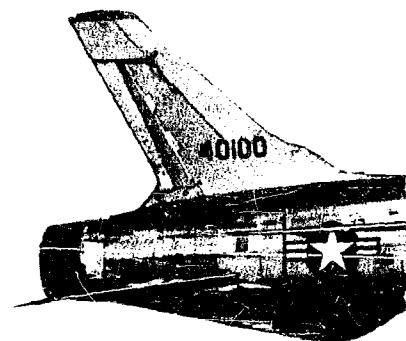
An adjustable stick damper limits the control stick rate in the stabilizer control system. The longitudinal control system also incorporates a mechanical advantage ratio changer, which provides variable ratios of control-stick-to-stabilizer sensitivity. Manual operation of the system allows the use of any mechanical advantage ratio from 1 to 1 to 3 to 1. Automatic operation of the system will be available at a later date.

An aileron-spoiler mechanical advantage changer is incorporated in the lateral control system. This changer reduces the ratio of aileron-spoiler travel to control stick deflection in proportion to the extension of the trailing edge flaps. As the aerodynamic effect of the spoiler increases with trailing edge flap extension, the ratio changer maintains a uniform roll response to a given stick deflection for all positions of the trailing edge flaps. The lateral control system incorporates an aileron lock-out motor, which renders the ailerons inoperative at speeds above 415 knots.

A mechanical advantage ratio shifter is incorporated in the rudder control system. The ratio of rudder deflection to rudder pedal movement is 1 to 1 until an airspeed of 210 knots is reached. At this point, the rudder control shifts to a 1 to 4 ratio.

Three hydraulic systems are provided in the airplane:

1. A dual primary system, powered by two engine-driven pumps, for the flight control systems.
2. A utility system powered by an air-turbine motor (fed from the engine compressor) for the landing gear, speed brakes, leading edge flaps, refueling probe, bomb bay doors, hydraulic gun drive, nose wheel steering, duct plugs and bleed doors, and the wheel brakes.



3. An emergency system powered by a ram-air driven pump to supply the primary No. 1 system in case of primary system pump failure.

Either primary system can power all flight control systems. The utility reservoir provides the fluid for both utility and emergency systems and the utility pump will augment the output of the emergency pump. Either the emergency system, or both emergency and utility systems can be used to operate the primary No. 1 system. During normal operation each primary system powers one side of each of the tandem power actuators in the flight control system as well as providing power to the control stick booster.

The following special conditions existed during the test program due to the stage of development.

1. The bomb bay doors were not opened in flight because of the fuel tank installation and test instrumentation lines in the bomb bay.
2. Automatic control of the mechanical advantage ratio shifter in the longitudinal control system was not available at the time of the test. Provisions for manual operation of the system were provided.
3. Gross weight at engine start was 37,900 pounds. This weight includes 350 gallons of fuel in the bomb bay tank.
4. Limit load factor was 7g in subsonic flight and 4.9g in supersonic flight. The limit indicated airspeed was 750 knots during the Phase II tests.
5. The secondary exhaust nozzle was in the afterburning position for all of the Phase II flights. At that time no provisions were installed for closing the nozzle to the optimum non-afterburning position.
6. The pilot's airspeed system was connected to the test nose boom. All indicated airspeeds shown in this report are therefore from the test nose boom.



TEST RESULTS

■ cockpit evaluation

The cockpit of the F-105B is unsatisfactory. Design deficiencies described in the following paragraphs are so numerous that the cockpit must be substantially redesigned before the aircraft can be considered satisfactory for operational use. Particular emphasis must be placed on simplicity and clarity in presentation of cockpit controls.

Cockpit entrance is made through use of a non-integral ladder which hangs over either side canopy rail. The canopy is closed or opened by lifting the canopy up or pulling it down, and is locked by means of a manually-operated over-center locking mechanism located on the left hand cockpit wall below the canopy rail. The locking mechanism interferes with the pilot's hand when moving the throttle between the cut-off and idle positions. No provisions are made for holding the canopy open at intermediate positions and it must remain closed during taxi operations. This results in discomfort during ground operation in hot weather.

Although no ejection tests have been performed on the F-105B seat, it will be completely inadequate to protect the pilot from the hazards of high speed ejection since the seat lacks stabilizing devices, limb retainers, a wind deflector, and a means of reducing deceleration. In addition, it is not known whether the

seat itself will withstand high speed ejection. This capability should be demonstrated in high speed track tests, particularly as seats of similar construction have failed structurally when ejected at supersonic speeds during early development tests.

The pilot's seat installation will not accommodate the use of a pressure suit, and anti-exposure and related personal equipment by average or large pilots. (The approximate width of the average pilot in complete anti-exposure gear is 28½ inches, but the seat is only 23 inches wide.) In consideration of the altitude capability of this airplane, the seat dimensions at the arm rests should be increased several inches to provide a reasonable degree of safety during ejection when wearing a pressure suit. In the present configuration it would be impossible to bring the elbows within the arm rest envelope during ejection. This could result in arm flailing, broken bones and related traumatic effects.

It is recommended that a crotch strap be provided on the seat installation, since severe vertebral injury can be sustained during a crash or hard landing if the pelvis is permitted to move forward. In addition to providing pelvic restraint, the crotch strap holds the lap belt down, which prevents loosening of the shoulder straps and forward movement of

the shoulders in a rapid deceleration. The crotch strap must be designed so that it will not interfere with pilot-seat separation during low altitude ejection.

The thick, cushioned seat filler used in the F-105B during flight test is a dummy survival kit which approximates the programmed kit. This dummy kit is unsatisfactory for use, since the thick cushion may cause or permit back injury during crash landings.

Rudder pedal adjustment cannot be made quickly and accurately. The detents for various rudder pedal positions should be numbered to facilitate pilot adjustment.

The throttle is too far forward in the afterburner range. An average size pilot can reach it with arm fully extended without releasing his shoulder harness lock, but the position is uncomfortable and tiring. Modifications to provide more comfortable throttle operation are recommended.

The present throttle quadrant contains a manual friction control lever. This is an additional cockpit control that could be eliminated. Installation of an automatic anti-creep spring assembly such as in the F-102A is a satisfactory arrangement that eliminates the need for manual friction control.

The aircraft is equipped with position indicators for both the trailing edge and leading edge flaps. The trailing edge selector handle is located outboard of the throttle and the leading edge selector handle is located inboard of the throttle. For production aircraft it would be desirable to have one handle, outboard of the throttle, for both flaps. The following detents are recommended:

1. Forward position (both flaps up).
2. Leading edge flap optimum position or positions.
3. Full aft position (leading edge and trailing edge flaps in take-off and landing position).

The need for positions other than those recommended is not anticipated; therefore, the flap position indicators can be eliminated.

The drag chute is operated by a standard drag chute control handle located on the upper left instrument panel. After the chute is deployed, the chute can be jettisoned by rotating the handle 90 degrees counterclockwise and pulling. If the pilot pulls the handle straight back to the stop and then rotates the handle counterclockwise (outboard) and pulls for

chute release the system works satisfactorily. However, there is a natural tendency to twist the wrist outward when pulling the arm toward the shoulder; consequently it is easy to jettison the chute inadvertently. One case of unintentional chute jettisoning occurred during Phase II tests. Inadvertent chute jettisoning resulting from a similar chute control handle deficiency constituted a serious safety problem in a tactical unit using another Century Series airplane. It is recommended that a detent override be installed between the deploy stop and the counter rotation position.

The handles for emergency brakes and emergency gear extension are labeled "PULL". Each handle should be labeled according to its function in addition to the plastic edge lighting identification provided.

A trim light is provided that indicates take-off trim for the longitudinal and lateral controls. A requirement has been established that Century Series fighters be equipped with a single light and a single button to trim for take-off position in all three axes. The switch and light should be located in the position now occupied by the present trim light.

In the test aircraft the rudder trim switch was located on the aft inboard section of the throttle quadrant. This has proven a desirable location and should be retained for production aircraft.

The present warning light configuration has a master warning light in the upper right side of the instrument panel and a long narrow warning identification panel under the right canopy edge extending aft of the pilot. This warning panel is difficult to see and is a distinct annoyance to the pilot when boost pump out and intermittent fuel low level lights require continual attention to the rear of the cockpit. This panel should be relocated and reshaped to fit on the right side forward instrument sub-panel similar to other Century Series fighters. The master warning light should be relabeled "Master Caution".

A red light for fire and overheat warning on the upper center of the forward instrument panel provides a steady light for fire and an intermittent light for overheat. A previous recommendation (YF-105A Phase II Report) for two fire warning lights, one on each side of the forward panel, is withdrawn for this airplane in the interest of cockpit standardization.

The present landing gear warning system includes a standard cockpit landing gear warning horn.

The warning horn should be replaced with a type MA-1 buzzer which gives an intermittent warning buzz in the headset.

The fuel quantity indicating system in this aircraft requires four separate gages to indicate fuel for internal, auxiliary bomb bay, and two external wing tanks. The fuel gage indicating system provided in the YF-105A used for Phase II tests incorporated all systems in a single gage and should be retained in production aircraft. The fuel low level warning light is actuated by the fuel gage, which is unsatisfactory. A float should be installed in the sump tank to provide reliable low level warning. The present gage selector for individual tanks is at the aft end of the right-hand console. This location is inconvenient to the pilot; the selector should be moved forward, adjacent to the gage. Provisions are made for a centerline fuel tank mounted beneath the bomb bay, but there is no fuel gage to indicate centerline fuel. Since this tank is capable of aerial refueling, fuel quantity indication should be incorporated in the single gage recommended.

At present, switches to turn the fuel boost pumps on and off are provided in the cockpit. The switches should be removed from the cockpit, since the pumps can be operated in a dry fuel tank and are never turned off in flight.

The circuit breakers in the cockpit are non-standard and pop in, rather than out, when the circuit overloads, making them hard to identify by feel and sight and very difficult to reset.

The mechanical advantage gage and selector switch for the rudder, located on the left forward side panel, should be removed, since the installation of a control which automatically changes the mechanical advantage of the rudder at 210 knots IAS makes both unnecessary.

The M-1 airspeed-Mach indicator is unsatisfactory, and should be replaced with an indicator which can be read more easily.

The pressure indicators for the No. 1 and No. 2 hydraulic systems fluctuate to such an extent that they are distracting to the pilot. The indicators should be damped.

Large variations in cabin temperature have been experienced with changes in airspeed, altitude and power settings. Excessive attention is required to arrive at a uniformly comfortable temperature. With the temperature control in a fixed position, the cockpit temperature increased from 40 degrees to 175

degrees F during an acceleration from 0.93 Mach number to maximum speed. It is recommended that the cabin temperature control system be made automatic and the control lever be replaced with a knob rheostat. The location of the temperature control on the left console should be retained.

The D-2 regulator is not satisfactory for pressure indications on liquid oxygen systems. The D-2 pressure scale is 0-500 psi, whereas the liquid system supplies oxygen under a pressure of 70 (± 5) psi. The result is a low reading on a gross scale, which is useless for proper determination of pressure or system malfunction. Further, there is no pressure suit operation provision incorporated in the D-2 regulator. The D-2 regulator should be replaced by an oxygen panel reading 0 to 150 psi, which incorporates provisions for a pressure suit.

The AN/ARC-34 command radio set permits selection of preset frequencies by rotation of the selector knob on the control panel on the right console. A numerical indication of the selected channel appears in a window forward of the selector knob. Thus, the pilot must look down and to the right to monitor channel selection and must remove his right hand from the stick to change channels. Aircraft accident statistics indicate that the process of changing radio frequencies in fighter aircraft involves a considerable hazard under low altitude IFR conditions. Since the F-105B may be required to operate under IFR, the channel indicator should be located on the instrument panel ahead of the pilot, and the channel selector should be located on the left console aft of the throttle. A command set remote indicator for this purpose is being tested by the Equipment Laboratory, Wright Air Development Center. The equipment bears the nomenclature ID-572/ARC Remote Preset Channel Indicator (Hayes Aircraft Company) for use with AN/ARC-34, 33 or 27 radio sets. It is recommended that this type equipment be installed in the F-105B as soon as practicable for Phase IV and VI flight testing.

■ starting, taxiing and ground handling

Starting procedure is similar to that in other Century Series fighters. A start switch is activated which admits starting air to turn the engine, and the throttle is advanced to the idle position at 12 percent rpm. Acceleration to idle rpm is smooth and rapid,

requiring approximately 20 seconds to reach an idle rpm of 59 percent. Starting characteristics are similar to the J57 engines, with maximum starting temperatures reaching approximately 350 degrees centigrade.

Power is advanced to 64 percent rpm to start the taxi roll. The throttle can then be reduced to idle, which provides sufficient thrust for normal taxi speeds. Visibility during ground operation is excellent. The nose wheel steering system is unacceptable due to slow response and overshoot tendencies. The system was considered unusable during these tests and differential braking was employed for ground control. The brake feel and effectiveness is satisfactory at taxi speeds. A distinct "gear walk" is encountered during all ground operation. It is particularly apparent to an observer outside the aircraft, and at speeds below 50 knots it can occasionally be felt by the pilot as a chattering in the rudder pedals when using wheel brakes. This gear flexing indicates the possibility of a structural fatigue problem area and should be completely investigated by the contractor. At present, taxiing must be done with the canopy closed, which will cause excessive pilot discomfort in hot weather.

■ take-off and initial climb

All take-offs are made with leading edge and trailing edge flaps fully extended. The brakes are adequate to hold the aircraft in military power. The brakes are released in military power and as ground roll begins the afterburner is lighted. Since nose wheel steering was not usable, differential braking was used to maintain directional control until a speed of approximately 60 knots IAS was obtained where the rudder became effective. The aircraft is rotated to take-off attitude between 140 and 150 knots and the take-off is made at 180 knots with either military or maximum power.

The gear is retracted immediately after the aircraft becomes airborne. During the retraction cycle an objectionable and unacceptable pitch down occurs. Gear retraction time is approximately 6 seconds. The trailing edge and leading edge flaps are retracted immediately after gear retraction. Very little trim change is encountered. At 210 knots IAS the rudder mechanical advantage shifts from a 1 to 1 to a 4 to 1 ratio. This shift results in a small directional trim change that requires re-trimming the rudder. This trim change should be eliminated. Take-off perform-

ance is presented in Figure 1, and optimum take-off speeds and distances are listed in the following table.

TAKE-OFF PERFORMANCE

Gross Weight at Engine Start 37,900 Pounds

Power	Ground Roll ft.	TAS at Take-Off kts.	Nose Boom IAS at Take-Off kts.	Total Distance Over 50 Foot Obstacle ft.	TAS at 50 Feet kts.
Maximum	3500	180	179	5,600	210
Military	5500	185	182	10,200	210

■ climbs

Climbs were made with maximum and military power using the contractor's recommended climb schedule. Climb performance at low and medium altitude is very good; however, the absolute subsonic ceiling of 47,000 feet with maximum power is lower than the contractor's estimate of 53,000 feet. The absolute ceiling using military power is 39,500 feet.

The acceleration to climb speed with maximum power is rapid and the aircraft must be rotated at approximately 0.82 indicated Mach number to arrive at the proper climb angle at the recommended climb schedule of 0.89 indicated Mach number. The climb attitude is steep initially, but decreases rapidly above 20,000 feet as the rate of climb decreases. A maximum power subsonic climb yields a combat ceiling of 46,500 feet. A decrease in engine pressure ratio and exhaust gas temperature was encountered at altitudes above 35,000 feet during the subsonic maximum power climbs. The fuel control appears to be responsible for this decay, and an investigation of fuel scheduling at altitudes above 35,000 feet is recommended. One supersonic climb, which was conducted above 35,000 feet at a constant Mach number of 1.77 resulted in the same rates of climb and combat ceiling as the subsonic climbs.

Climb performance is shown in Figures 2 and 3 and maximum power climbs are summarized in the following table for a gross weight of 37,900 pounds at engine start.

Altitude ft.	Rate of Climb ft./min.	Time min.	Fuel Used lbs.	TAS kts.
Sea level	—	0	0	640
10,000	25,000	0.3	330	610
20,000	18,700	0.65	680	580
30,000	12,000	1.3	1040	550
40,000	4,700	2.5	1480	535
45,000	1,200	4.4	2000	535

Altitudes above the steady state ceiling can be achieved by accelerating to high supersonic speeds at 35,000 feet and then zooming to higher altitudes. One zoom climb was conducted starting from a speed of 1.9 Mach number at 35,000 feet. A constant Mach number (1.9) climb was made to 40,000 feet, where a 1.7g pull-up was initiated. The pull-up was held until the Mach number bled off to 1.6; a constant attitude was held from this point. The afterburner was turned off at 60,000 feet to prevent an afterburner blow-out and possible engine overspeed. A slow push-over was started at 60,000 feet which resulted in leveling out at 63,000 feet at a Mach number of 1.27. This climb is presented in Figure 18.

■ level flight

The level flight acceleration performance of the F-105B is not impressive when compared with the F-104 or F-107. The time required to accelerate to maximum speed and the fuel used during acceleration will severely limit the tactical use of the maximum speed capability.

The performance of this airplane varies widely with variations of ambient temperature. A variation in ambient temperature of 10 degrees C will change the maximum speed approximately 0.15 Mach number at 35,000 feet and approximately 0.12 Mach number at 20,000 feet. The effect of ambient temperature on acceleration performance is shown in Figures 6 through 8. The level acceleration performance at 35,000 feet under standard conditions, utilizing inlet duct bleed door schedule "F", is as follows:

Mach No.	Time min.	Fuel Used lbs.	Fuel Flow lbs/hr.	Distance Traveled n. m.
1.0	0	0	24,000	0
1.2	0.9	410	28,300	10
1.4	1.85	790	32,500	21
1.6	2.90	1500	36,800	37
1.8	4.35	2450	40,500	61
1.95	9.0	5500	42,800	140

The maximum speed of the airplane was measured at 20,000 feet where a Mach number of 1.49 was obtained, which is close to the temporary limit indicated airspeed of 750 knots. The maximum speed

was measured at 35,000 feet, using bleed door schedule "C", and a Mach number of 1.89 was attained. A Mach number of 1.95 was attained on a later flight using a different engine and utilizing bleed door schedule "F".

Although a Mach number of 1.95 can be attained on a standard day, fuel reserves necessary for returning to base and landing will dictate extremely short flight duration at that speed. Practical use of the airplane will be limited to speeds below 1.8 Mach number and careful flight planning will be required to use this speed capability when a bomb bay fuel tank is not carried. Acceleration data is shown in Figures 6 through 9.

At the start of the Phase II tests, bleed door schedule "C" (Figure 32) was used in conjunction with the duct plug schedule shown in Figure 31. Using this schedule the bleed doors start to open at 1.34 Mach number on a standard day and reach an opening of 17.3 degrees at Mach 2. With development of the inlet system, schedule "F" (Figure 33) was inaugurated prior to flight No. 11. This schedule was identical to the previous one up to Mach 1.76. Above this speed, schedule "F" maintains an essentially constant bleed door opening.

The specific range of the F-105B is considerably lower than the contractor's estimates. At 20,000 feet and 33,000 pounds gross weight, a specific range of 0.114 nautical air miles per pound of fuel is available at the optimum cruise speed of 0.775 Mach number. This is approximately 13 percent less than estimated. At 35,000 feet altitude and 32,000 pounds gross weight, a specific range of 0.134 nautical air miles per pound of fuel (approximately 20 percent less than estimated) is available at the optimum cruise speed of 0.92 Mach number. Leading edge flaps are 30 percent extended at all subsonic speeds on the contractor's recommendation. It should be noted that the secondary nozzle formed by the speed brakes was not in optimum cruise position during these tests. An improvement in efficiency and in cruise can be expected with optimizing of the secondary nozzle. Specific range data is presented in Figure 15.

Level flight thrust required at 20,000 and 35,000 feet is presented in Figures 11, 13 and 14. The subsonic data is representative of the aircraft with 30 percent leading edge flaps and afterburner nozzles closed. The supersonic data represents the aircraft with all flaps retracted, afterburner nozzles wide open, and inlet duct bleed doors operating above

1.35 Mach number. The thrust required at supersonic speeds was obtained by subtracting excess thrust (obtained from measuring acceleration) from measured thrust available.

■ turning performance

The steady state turning performance at supersonic speeds is thrust limited to load factors much smaller than the lift limited capability of the airplane. Sufficient thrust is available to maintain a steady state load factor of 2.25g at 1.2 Mach number and 1.8g at 1.7 Mach number in a level turn at 35,000 feet at a gross weight of 32,000 pounds. This data is presented in Figure 17.

The lift-limited maneuvering capability of the airplane is good at supersonic speeds; however, like other Century Series fighters, the subsonic maneuvering capability is low. Maneuvering tests were conducted at supersonic speeds up to the Phase II limit of 4.9g with no indication of buffet. At subsonic speeds at 35,000 feet, buffet was encountered at 2.8g at 0.93 Mach number. At 40,000 feet buffet was obtained at 2.6g at 0.98 Mach number. The load factor available in the power approach configuration is limited by heavy buffet to 1.9g at 0.47 Mach number and 20,000 feet.

■ stalls

The handling characteristics of the airplane are good in stall approaches. However, complete stalls, as well as spins and spin recoveries, have not been demonstrated. Difficulties with other Century Series fighters in spin recovery point up a requirement for an early spin program.

Unaccelerated stall approaches were performed at 20,000 and 35,000 feet in the clean configuration with leading edge flaps at 30 percent. Initial buffet was observed at 215 knots IAS and the buffet increased slightly as airspeed decreased. Buffeting became heavy at 155 knots and lateral control began to deteriorate. The stall was discontinued at 145 knots, which was the lowest speed demonstrated by the contractor. The longitudinal stability is positive to minimum speed and the control characteristics are good.

An unaccelerated stall approach was performed at 20,000 feet in the power approach configuration. Initial buffet was observed at 190 knots IAS. Buffet intensity was light above 155 knots. Heavy buffet

was observed at 155 knots and lateral control began to deteriorate. Heavy buffet continued to 139 knots, where the stall was discontinued. The longitudinal stability is positive to minimum speed and control response is good.

Accelerated stalls were accomplished in conjunction with maneuvering flight tests. In both cruise and power approach configurations acceleration was increased until heavy buffet and a roll-off occurred. The roll-off was gradual with a low roll rate and never offered difficulty in control. The roll-off stall characteristic is easy to avoid as an adequate buffet margin exists between initial and heavy buffet. Accelerated maneuvers at supersonic speeds were made to the temporary load factor limit of 4.9g without encountering buffet.

■ descents

Satisfactory let downs can be made by reducing power to idle rpm. Power reduction below military power at speeds above 1.7 Mach number was prohibited by the contractor due to uncertainty of stable duct operation. This restriction should be removed. However, speed brake extension provides faster deceleration and more rapid descent rates than power reduction. There is no trim change associated with speed brake extension. Deceleration with speed brake extension is greater than that obtained with any production USAF fighter to date. A time history of a speed brake opening in level flight is shown in Figure No. 75. One attempt was made to maintain a 0.9 Mach number descent but the descent became nearly vertical at 20,000 feet. A constant airspeed descent is more practical since the variation in pitch angle is smaller.

■ landing

The handling characteristics of the airplane are good in the traffic pattern and during landing. Landing distances compare favorably with other Century Series fighters.

During the Phase II tests very little tire wear was encountered despite the use of moderate to heavy braking. It was not possible to skid a tire, even with heavy braking, which indicates that adequate braking power is not available to the pilot. The contractor should investigate the adequacy of the F-105B wheel brakes, and improve braking power to minimize landing distance. The landing speed and

weight of the F-105B require excellent brakes and a landing drag chute for suitable operation on existing runways. To insure the maximum use of wheel brakes without failure or damage to tires, wheels and landing gear struts, anti-skid devices should be incorporated in the wheel brake system.

A speed brake configuration was provided which allowed use of only the side speed brake segments when the landing gear is lowered. The top speed brake is closed to allow drag chute deployment and the bottom brake is closed because of proximity to the ground on landing. This arrangement is satisfactory and should be provided on production aircraft.

A large reduction in power must be made to decrease airspeed to the landing gear limit speed of 243 knots IAS. The large increase in drag and sink rate which accompanies gear extension requires rapid movement of the throttle to nearly full power to prevent a dangerous reduction in traffic pattern airspeeds. To prevent awkward and undesirable techniques at traffic pattern speeds it is recommended the contractor raise the landing gear limit airspeed at least 15 knots.

The initial approach is made at 350 knots indicated airspeed. Leading edge flaps were fully extended on the initial approach to provide improved low speed maneuvering and to reduce procedures in the traffic pattern. Speed brakes may be used on the "break", but are not required as power reduction decreases speed to allow arriving on the downwind leg at 240 knots. The landing gear is lowered on the downwind leg. Trailing edge flaps are lowered at 220 knots and the base leg is flown at 200 knots. The turn to final approach is made at not less than 190 knots, and a speed of 180 knots is recommended for final approach. The landing flare and touchdown are typical of other Century Series fighters. Power is reduced to idle in the flare and touchdown speeds range from 150 knots to 165 knots. The drag chute is deployed immediately after touchdown, and the nose is lowered to obtain directional control. A slight delay (less than 2 seconds) is required for the drag chute to fully deploy. Braking may be used to stop the aircraft as soon as the nose wheel is on the runway.

LANDING PERFORMANCE

Sea Level Standard Day No Wind Conditions

	Ground Roll ft.	Distance Over 50 Foot Obstacle ft.	TAS at 50 Feet kts.	TAS at Touchdown kts.	Nose Boom IAS at Touchdown kts.	Gross Weight lbs.
Drag Chute Deployed	3760	6,320	179.5	147.5	28,800
at Touchdown —	3815	6,235	184.5	154	155	29,600
Moderate Braking	3305	5,335	167	149	156	28,700
	2835	4,985	162.5	142	28,900
	4115	4,940	174	168	157	28,600
	4280	6,800	189	167	157	29,000
	3430	5,580	181	165.5	155	29,500
No Drag Chute —	8420	10,240	183.5	166.5	162	28,700
Moderate Braking	7450	9,090	181.5	169	160	28,800
	6140	8,340	186	160	160	29,000
	6120	8,275	175	159	155	29,300
	6090	8,855	170.5	161	160	29,800

■ dead stick landings

Dead stick landings with the engine flamed out have not been demonstrated by the contractor. Experience with the F-101 and F-104 aircraft indicates that flame-out landings are questionable and should be demonstrated early in the test program to develop satisfactory techniques.

■ airspeed system calibration

The standard wing boom airspeed system has good characteristics in the subsonic speed range, the position error being essentially zero. The calibration in the supersonic speed range, however, shows the erratic position error typical of wing boom systems. Very poor repeatability was obtained in the super-

sonic calibration. The maximum position error is 0.1 Mach number and approximately .05 Mach number scatter was obtained in the region between Mach 1 and 1.6. This wing boom installation is unsatisfactory for supersonic indications.

The test nose boom shows the conventional nose boom characteristics. The position error is linear in the subsonic region and zero in the supersonic speed range. The maximum error is 0.035 Mach number in the transonic range.

The airspeed systems were calibrated at 20,000 and 35,000 feet altitudes. The subsonic calibration was obtained by flying stabilized points in formation with an F-86 pacer aircraft. The supersonic calibration to 1.3 Mach number was obtained by having a pacer establish a constant altitude smoke trail and accelerating the F-105 along this trail. The supersonic position error for the nose boom was found to be zero up to 1.3 Mach number and a zero position error above this speed was assumed. Supersonic data was also obtained for the wing boom system by cross plotting the data for the two systems, utilizing the calibration already established for the nose boom. Airspeed calibrations are shown in Figures 22 and 23.

■ control system

The lateral and longitudinal control systems in the F-105B are unsatisfactory. The lateral control is deficient because excessive stick movement around neutral is required before airplane response is noted. This is particularly annoying in formation flight when continual stick movement is required to maintain or correct bank angles. The result is a continuous wing rocking and a lack of positive control. The longitudinal control is too sensitive with the pitch damper off. Small, short period oscillations which cannot be satisfactorily damped are easily induced. This makes it difficult to apply and maintain positive g forces. These deficiencies should be corrected to provide good control about all axes for the precise requirements in LABS maneuvers and other bomb delivery operations, and ease of tracking for gun laying operations against ground or airborne targets.

The inclusion of the pitch damper improves longitudinal control, which is considered satisfactory with the damper operating. The longitudinal break-out force should be reduced to comply with specification requirements. Rudder break-out force and gradient are considered satisfactory, even though

break-out forces do not meet specification requirements. Break-out forces in pounds were as follows:

	Forces Measured	Limits — MIL-F-8785 (ASG)
Longitudinal	4	3
Lateral	2	2
Directional	19	7

The control deflection calibrations are presented in Figures 76 through 84.

■ dynamic stability

The dynamic longitudinal and lateral directional stability of the airplane with pitch and yaw dampers on is satisfactory. The longitudinal damping ratios are between 0.4 and 0.8 of critical damping (0.8 to 0.3 cycles to damp to 0.1 amplitude) and meet the requirements of specification MIL-F-8785 (ASG). However, both the dynamic longitudinal and lateral directional stability are unsatisfactory with stability augmentation off. The data for the F-105B airplane with stability augmentation off is essentially the same as the data obtained for the YF-105A airplane as presented in Phase II report AFFTC-TR-56-18. The dynamic longitudinal stability with dampers off does not meet specification requirements. Longitudinal control is extremely sensitive with stability augmentation off, which makes arriving at and maintaining a desired pitch attitude very difficult. Short period oscillations which cannot be damped satisfactorily are easily induced. Dynamic stability data is shown in Figures 45 through 51 and 58 through 65.

■ lateral control

The roll capabilities of the airplane are considered poor. The maximum roll rate obtained was 150 degrees per second at 0.94 Mach number at 35,000 feet.

Full control deflections were not obtained during the Phase II tests. The lateral stick travel is excessive and the control stick hits the pilot's leg before full deflection is obtained. Several rolls were made in which the pilot moved his leg in order to reach the lateral stick stop, but at full stick deflection full spoiler deflection was not obtained. Control stick travel should be reduced and full spoiler deflection should be made available to improve rolling performance and to allow the pilot to obtain the maximum rolling capability of the airplane. Roll tests were made at 0.86, 0.94 and 1.4 Mach numbers at 35,000 feet and at 1.4 Mach number at 45,000 feet. Approx-

mately 6 degrees of sideslip were obtained at 0.94 Mach number and 3 degrees of sideslip at 1.4 Mach number in rolls at 35,000 feet. The rolls at 1.4 Mach number at 35,000 feet were also accomplished from a 2g trim condition. The sideslip angles obtained were approximately the same as for a 1g entry. Rolls at 1.4 Mach number at 45,000 feet resulted in sideslip angles of 2 degrees.

Inertial coupling was not a problem at the conditions tested; however, a more comprehensive program should be conducted to assure freedom from serious roll coupling effects throughout the speed and altitude envelope of the aircraft especially if roll capability of the aircraft is increased. Roll data is presented in Figures 66 through 74.

Excessive lateral trim is required when accelerating in the supersonic speed region and opposite trim is required when decelerating through the same speed range. This trim change is objectionable and should be eliminated.

■ static longitudinal stability

The stabilizer position variation with speed is satisfactory (requires forward stick for increasing airspeed) except in the transonic region. A trim change occurs between 0.9 and 1.0 Mach numbers. This trim change is unacceptable in magnitude and abruptness. It occurs in a region that will be used extensively for cruising, and will be an annoyance in pilot control. A Mach sensing device could be incorporated in the longitudinal control to provide a stable stick force gradient throughout the transonic speed regime. Trim curves are shown in Figure 34.

■ maneuvering flight

The maneuvering flight characteristics are considered satisfactory in most of the areas investigated; however, the initial stick force gradient should be reduced and the stick force gradient made linear. In addition, the longitudinal break-out force should be reduced to comply with specification requirements.

The stabilizer angle and stick force required in maneuvering flight increase with an increase in normal acceleration up to the stall buffet, or to the maximum accelerations obtained. The stick force gradient is non-linear with acceleration and the initial gradient from trim is higher than desirable. The average force gradient, however, meets the requirements of MIL-F-8785(ASG) with two exceptions. The stick force gradient at .98 Mach number at 40,000 feet does not

meet specification requirements in that the average force gradient is too high and the local variation from the average gradient is greater than 50 percent. The second exception is the stick force gradient in the power approach configuration at 20,000 feet, which is higher than specification requirements. The maneuvering capability at subsonic speeds is limited by airframe buffet, but maneuvering tests were conducted at supersonic speeds up to the Phase II limit of 4.9g with no indication of buffet. The maneuvering data is shown in Figures 35 through 44.

■ LABS maneuvers and dive bombing

The F-105B can satisfactorily accomplish dive bombing and LABS maneuvers with stability augmentation operating. Both maneuvers are more difficult to control precisely with stability augmentation off. The bomb bay doors do alter the aircraft trim either when they are cycled or when speed is changed with the doors open. An airframe buffet is encountered with the bomb bay doors open at indicated airspeeds greater than 350 knots. This buffet increases in intensity as speed is increased to 450 knots, then remains constant at higher speeds. This buffet is not objectionable.

Two simulated dive bombing runs and two simulated LABS maneuvers were accomplished with F-105B USAF S/N 54-101. Each maneuver was accomplished with and without stability augmentation and one dive bombing run was made with speed brakes extended. All maneuvers were accomplished satisfactorily except for the difficulties with precise control when stability augmentation was not used. Some pieces of skin on the ventral fin, speed brakes and wings were lost during maneuvers as described under "Structural Failures". The LABS maneuvers were made with an afterburning entry at 5000 feet and 575 knots IAS. The maneuvers resulted in an altitude gain of 13,000 feet with an initial 4g pull-up and a completion speed of 220 knots.

■ night flight

One night flight was made on F-105B S/N 54-101 to evaluate cockpit lighting. An F-100 was utilized to fly formation on the F-105B to observe position and formation lighting. Lighting was considered generally satisfactory in control and intensity variations. The deficiencies noted can be corrected with minimum effort.

The cockpit lighting must be improved in several areas. The throttle quadrant has no identification lighting. This includes flap and rudder trim switch identification. The refueling probe and emergency belly tank release handles are adjacent to one another but have no lighted identification. The written identification adjacent to the master caution light that reads, "MASTER CAUTION AND PRESS TO RESET", is not lighted. The light on the fuel flow indicator does not illuminate the gage between the two and six o'clock positions. This includes the fuel flow range between 2000 and 5000 pounds per hour, a range that requires accurate monitoring during cruising flight.

The fire control panel contains two dials, one for IP range, and one for burst height. Each has a separate red light for night identification of the dial readings. The light reflection from the dials is excessive. Regardless of cockpit lighting intensity selections, a red reflection is apparent on the forward canopy, extending around the canopy in a ring between each canopy side rail. The circuit breaker panel on the left hand console has excellent identification lighting; however, greater facility would be offered in readability if the circuit breaker wording were rotated 90 degrees counterclockwise to be in line with the longitudinal axis of the airplane.

The position and formation lighting arrangement is satisfactory. With exterior lights in the dim position the intensity is ideal for formation flight with two exceptions. The intensity of the white light on the bottom of the forward fuselage and the amber light high on the tail should be reduced to a level comparable with the other fuselage lights. The taxi light is adequate in side and forward coverage, but it is too dim. An intensity increase of at least 100 percent is necessary to make this light adequate for normal taxi operation. The landing lights offer excellent intensity and coverage, but shine straight ahead of the aircraft during a normal landing approach. Proper ground illumination is not achieved until touchdown occurs. The beam should be directed downward sufficiently to provide ground illumination prior to the landing flare.

■ emergency gear extension

The present landing gear extension procedure is unsatisfactory. Two distinctly separate procedures may be used, depending upon the pilot's interpretation of the degree of utility hydraulic system mal-

function or failure. The normal procedure for lowering the gear, if utility pressure is greater than 2000 psi, is to place the gear handle down. If the gear does not come down, the emergency gear release handle must be actuated, which mechanically releases the gear uplocks, permitting the gear to drop and lock. If the gear remains retracted it is necessary to actuate the pressure dump valve switch. If the utility pressure is less than 2000 psi the emergency gear extension lever should be actuated prior to placing the gear handle down, followed by pressure dump switch actuation. Emergency gear extension is an unduly complex procedure and is an excessive requirement on the pilot in an emergency. Emergency gear extension should consist of placing the gear handle down, then actuating the emergency gear release handle.

■ structural failures

Several structural failures occurred in Phase II on three separate flights. The boundary layer ramp adjacent to the inlet duct failed and entered the engine intake duct during a test at 1.52 Mach number at 20,000 feet. Damage to the duct and engine was extensive. A similar flight caused failure of the skin on the ventral fin, and a flight to evaluate bombing maneuvers resulted in failure and loss of approximately one-third of the ventral fin, several small pieces contained in the speed brake construction, and the plastic filler used on the underside of the wing around the main spar. These failures cannot be tolerated, particularly since structural limits were never exceeded. Structural rework is mandatory in the areas mentioned.

■ inlet duct sucker doors

The present engine intake system on the F-105B includes air intake ducts in the landing gear wheel wells to augment the wing root intake ducts for ground operation. A contractor flight that resulted in a wheels-up landing revealed that the ducts caused sufficient pressure differential on the wheel well doors at low speed and high power settings to prevent gear extension with either normal or emergency extension procedure. This condition can occur at any time when flying below gear limit speed with the speed brakes open, which requires a high power setting. The landing gear will not extend until power is reduced. This deficiency is not acceptable and should be corrected so landing gear extension is completely independent of engine power settings.



CONCLUSIONS

The F-105B is potentially an excellent fighter-bomber; however, a large number of improvements are necessary before the airplane can be considered acceptable for operational use.

The airplane is capable of 1.95 Mach number in level flight at 35,000 feet and 1.49 Mach number at 20,000 feet under standard day conditions. However, Mach numbers above 1.8 are not considered usable because of the poor acceleration characteristics. Fuel reserves are nearly depleted by the time maximum speed is reached. Almost 9 minutes are required to accelerate from Mach 1 to Mach 1.95 at 35,000 feet. The combat maximum power ceiling is approximately 46,500 feet, but higher altitudes can be reached using zoom climb techniques. An altitude of 63,000 feet was reached at 1.27 Mach number from a zoom climb initiated from 40,000 feet and 1.9 Mach number.

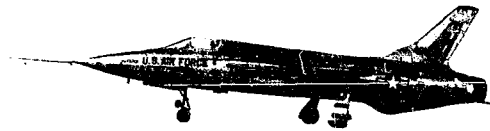
The lateral and longitudinal flight control systems are unacceptable. The lateral control response to stick movement near neutral is poor and the longitudinal control is too sensitive when the pitch damper is not engaged. These deficiencies make formation

flying difficult and compromise the ease with which a pilot can perform the mission requirements of the airplane. The fact that excellent control systems are possible within the present "state-of-the-art" of industry should make it mandatory that a better flight control system be provided in this airplane.

The present cockpit is unsatisfactory. The number of deficiencies will require a major redesign before the aircraft can be considered acceptable. The ejection seat is inadequate for bail-out at high Mach numbers, and has not undergone high speed track tests to indicate deficiencies in areas where pilot escape is considered practical.

Several other areas require improvement and investigation prior to delivery to tactical organizations. Improvement should be made in the rolling performance and in the specific range. Satisfactory demonstrations of stalls, spins, intertial coupling, and dead stick landings are required.

Maximum priority should be given to the early incorporation of development fixes and continuing flight checks must be made to insure that fixes are satisfactory.



RECOMMENDATIONS

It is recommended that immediate action be taken on the following items and that they be accomplished prior to the release of any production airplanes:

1. Improve the lateral control system to provide better aircraft response to stick movement.
2. Improve the longitudinal control characteristics to provide better formation and tracking capabilities.
3. Reduce the magnitude and abruptness of the pitch that accompanies landing gear operation.
4. Demonstrate stall, spin and spin recovery characteristics.
5. Demonstrate dead stick landing capability and characteristics.
6. Fully investigate inertial coupling to determine if a problem area exists.
7. Conduct an immediate cruise power investigation.
8. Improve the acceleration performance to decrease the time and fuel required to accelerate to maximum speed.
9. Investigate fuel scheduling at altitudes above 35,000 feet to improve the airplane performance and increase the ceiling.
10. Accelerate development of the engine air intake system to remove the restriction on throttle reduction at high Mach number.
11. Provide landing gear extension that is independent of engine power setting.
12. Provide greater structural integrity of the air intake-boundary layer splitter to prevent failure similar to that encountered in Phase II flight tests.
13. Provide greater structural integrity of the ventral fin and speed brakes.
14. Demonstrate the structural integrity and stability of the ejection seat by high speed track tests.
15. Investigate the possible structural problem area due to excessive landing gear flexing and chatter.
16. Provide more braking power and incorporate anti-skid provisions.
17. Improve the nose wheel steering system so that it is usable for ground handling without the use of wheel brakes.
18. Make the side speed brakes available for use when the landing gear is extended on production airplanes.
19. Modify the emergency landing gear system so that emergency procedure is simplified to the following steps:
 - a. Landing gear handle down.
 - b. Pull emergency release.
20. Increase the landing gear limit extension speed at least 15 knots.
21. Reduce the longitudinal break-out force and the initial stick force gradient, and provide a linear stick force gradient.
22. Eliminate the longitudinal trim change that occurs in the transonic region.
23. Eliminate the excessive lateral trim required when accelerating or decelerating in the supersonic region.
24. Reduce the control stick travel required for full lateral control deflection.
25. Improve the lateral control system to provide full spoiler deflection with full control stick deflection and provide increased roll capability.

26. Eliminate the directional trim change that occurs when the rudder mechanical advantage shifts at 210 knots IAS.

27. Provide an airspeed system for production aircraft that has a more consistent position error than the present wing boom system.

28. Replace the present cockpit temperature control with an automatic control. Retain this control in the position presently used.

29. Modify the leading edge and trailing edge flap control system to provide operation by one handle, and eliminate the flap position gage.

30. Make throttle operation more comfortable by reducing the distance the pilot must reach to operate the throttle in the full forward position.

31. Provide an automatic "anti-creep" spring assembly in the throttle quadrant and remove the manual friction lock.

32. Provide a detent in the drag chute handle between the "Deploy" and the "Jettison" positions.

33. Replace the present low level fuel warning system with a float type low level system with a warning set for 1400 pounds.

34. Relocate the fuel quantity selector switch to a forward position adjacent to the fuel quantity gage.

35. Replace the four individual fuel gages with one gage which incorporates a fuel quantity indication for the centerline external tanks.

36. Provide a warning light panel similar to those in other Century Series aircraft, and place it on the right forward side panel.

37. The "Master Warning Light" should be relabeled "Master Caution."

38. Replace the landing gear warning horn with a warning buzz in the AIC-10 intercom.

39. Provide standard circuit breakers; i.e., breakers which "push-in" to engage and "pop-out" for overload.

40. Provide an oxygen pressure indicator with a 150 psi scale and provide a regulator to serve the pilot's pressure suit from the airplane system.

41. Modify the canopy design to allow taxi operation with the canopy partly open.

42. Relocate the canopy lock so that it does not interfere with the pilot's hand when moving the throttle between cut-off and idle positions.

43. Provide a single button to obtain rudder, stabilizer and aileron take-off trim, with a green light for trim indication.

44. Locate the directional trim switch on the inboard side of the throttle quadrant similar to the position used in the Phase II airplane.

45. Label the landing gear and emergency brake handles as to function.

46. Provide a remote channel indicator for the AN/ARC-34 radio and relocate the channel selector to the left console aft of the throttle.

47. Provide a more readable airspeed-Mach indicator.

48. Provide damping for the hydraulic system pressure gages to prevent distracting needle fluctuations.

49. Eliminate the rudder mechanical advantage indicator switch, and provide a light on the warning panel to indicate malfunction of the mechanical advantage changer.

50. Correct the night lighting deficiencies discussed in this report.

APPENDIX I

data analysis methods

■ take-offs

Take-offs were recorded with the AFFTC photo theodolite installation and were corrected to standard day, sea level, no wind conditions by the use of equations (6.301), (6.319), (6.321) and (6.311) contained in the Flight Test Engineering Manual, AF Technical Report No. 6273. The pressure ratio used to calculate test gross thrust for both maximum and military power take-offs was obtained at brake release with the engine operating at military power. Standard day gross thrust was obtained from the static thrust calibration at the standard temperature pressure-ratio.

■ climbs

Continuous climbs were flown and the data recorded every 1000 feet. This data was corrected to a standard condition of -55 degrees C above 35,332 feet by means of equations (5.203), (5.204), (5.205) and (5.206) contained in AF Technical Report No. 6273.

The increment of thrust to correct to standard thrust was obtained from an increment of P_{t7}/P_{t2} as explained in the paragraph on "Power Available". No weight corrections were made. Static source lag corrections were made to the indicated pressure altitude by the method outlined in AF Technical Report No. 6273. The static pressure lag constant was determined from ground tests and was found to be $\lambda_{s.L.} = 1.292$.

■ accelerations

Level flight accelerations were reduced by the following expression:

$$\begin{aligned} R/C_s = \frac{60}{\Delta t} \left[\frac{V_{T8} (\Delta V_{Tt} + \Delta V_w) + \Delta H_C \sqrt{\frac{T_{at}}{T_{as}}}}{g} \right] \\ + \Delta R/C_1 + \Delta R/C_3 + \Delta R/C_{power} \sim \text{ft/Min} \end{aligned}$$

Excess thrust was computed by the following expression:

$$Tex = \frac{R/C_s \times W_s}{101.333 \times V_{T8}}$$

The following notation applies:

R/C	corrected rate of climb
Tex	thrust available minus thrust required for level flight
Δt	time increment
V_T	true airspeed
ΔV_T	true airspeed increment
ΔV_w	wind velocity increment
ΔH_C	pressure altitude increment
T_{at}	absolute ambient temperature, test
T_{as}	absolute ambient temperature, standard
P_a	absolute ambient pressure
M	Mach number
b	airplane wing span
e	airplane efficiency factor
n	load factor
ΔF_n	standard net thrust minus test net thrust

■ level flight

Stabilized speed-power data was obtained at a constant weight-pressure ratio by flying successive points at higher altitudes to compensate for the weight reduction with fuel consumed. This technique minimized the correction of the data obtained. This data is presented with the engine pressure ratio as the power parameter since this pressure ratio is more sensitive to power changes than engine speed. The power available data shown in Figure 11 through 14 was obtained from level flight accelerations. The net thrust required in the supersonic regime was deduced from the maximum power accelerations by

subtracting excess thrust from standard day net thrust available. No attempt was made to correct the airplane drag for changes in inlet bleed door position resulting in changes from test to standard ambient temperatures.

In-flight drag computations were accomplished utilizing Pratt and Whitney's gross thrust coefficients shown in Figures 28 and 29. The thrust coefficients obtained from the thrust calibration shown in Figure 26 were used to check Pratt and Whitney's predicted values in the low pressure ratio region. Net thrust used in this report is defined as gross thrust minus ram drag. Engine airflow for the ram drag computations was obtained from Pratt and Whitney's curves for test and standard conditions. The ram drag expressions used are developed in Air Force Technical Report No. 6273 "Flight Test Engineering Manual". The drag polar shown in Figure 16 was obtained from stabilized level flight data.

■ power available

The power available on a standard day was obtained with the use of a pressure ratio bias curve (P_{t7}/P_{t2} versus T_{t7}). Power available was obtained by entering this curve at standard temperature corrected for adiabatic heat rise to obtain a standard day pressure ratio. The pressure ratio was converted to P_{t7}/P_a and the primary gross thrust was obtained by use of the gross thrust coefficients and the equation shown in Figures 28 and 29.

Primary net thrust was obtained by subtracting ram drag from the primary gross thrust. Secondary gross thrust was obtained by total pressure measurement in the ejector. Secondary ram drag was computed from pressure measurements in the fin duct. Secondary net thrust was obtained by subtracting ram drag from secondary gross thrust. Net thrust as used in this report is the total of primary net thrust plus secondary net thrust. No attempt was made to correct secondary thrust for temperature changes from test to standard day.

Pratt and Whitney representatives have indicated that the gross thrust coefficient curves in use at the time of this test may be 3 percent to 5 percent optimistic and that revised curves will be available in the near future. However, the in-flight thrust data in this report is based on the best information available at the time of the test.

■ fuel flow

The first 12 flights of the Phase II program were conducted with engine S/N 610016 installed and the remainder of the flights were accomplished with engine S/N 610023. The fuel flow data for the two engines, when corrected to inlet conditions does not agree. The reason for the disagreement in corrected fuel flow was not determined. The corrected fuel flows for the two engines are presented separately in Figures 20 and 21. These two plots were used to correct the respective data to standard day conditions by an increment of pressure ratio from test to standard day.

■ turning performance

The steady state turning capability of the airplane was obtained from maximum power accelerations and decelerations at 35,000 feet altitude. The accelerations were made at 1g normal acceleration. The decelerations were made at maximum power in a constant altitude turn by maintaining sufficient normal acceleration to decelerate the airplane through the supersonic speed range. A working plot of excess thrust versus Mach number was prepared and cross plotted to obtain a plot of $\left(\frac{nw}{M\delta}\right)^2$ versus excess thrust. The steady state turning capability was interpolated from the points of zero excess thrust on this plot.

■ landings

Landings were recorded with AFFTC photo theodolite installation. This data was corrected to standard sea level conditions with equations (6.403) and (6.406) contained in AF Technical Report No. 6273. No weight corrections were made.

■ temperatures

The recovery factor of the resistance type outside air temperature probe was found to be approximately 0.98 under stabilized conditions. The ambient temperatures in climbs and accelerations were obtained from weather balloon surveys to eliminate the lag encountered with the resistance type probe.

performance and stability plots

Figure No.

1	Take-Off Performance	19
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FIG. NO. 1
 TAKE-OFF PERFORMANCE
 F-105B NSAF NO. 24-100
 YJ-15-P-3 ENGINE
 NO EXTERNAL STORES
 GROSS WEIGHT 31,500 LBS.
 LANDING AND TRAILING EDGE FLAPS DOWN

○ MAXIMUM POWER
 △ MILITARY POWER

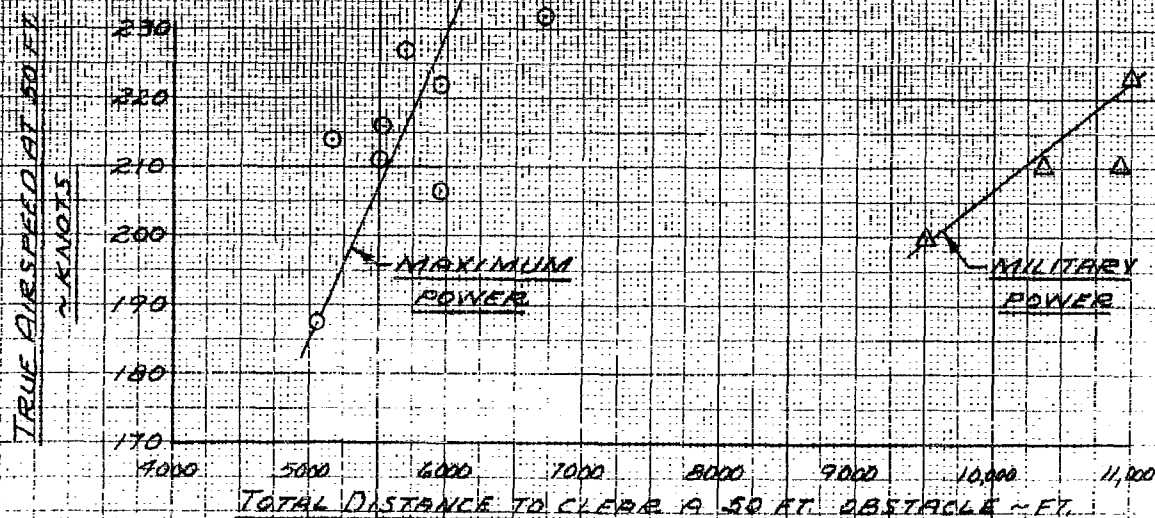
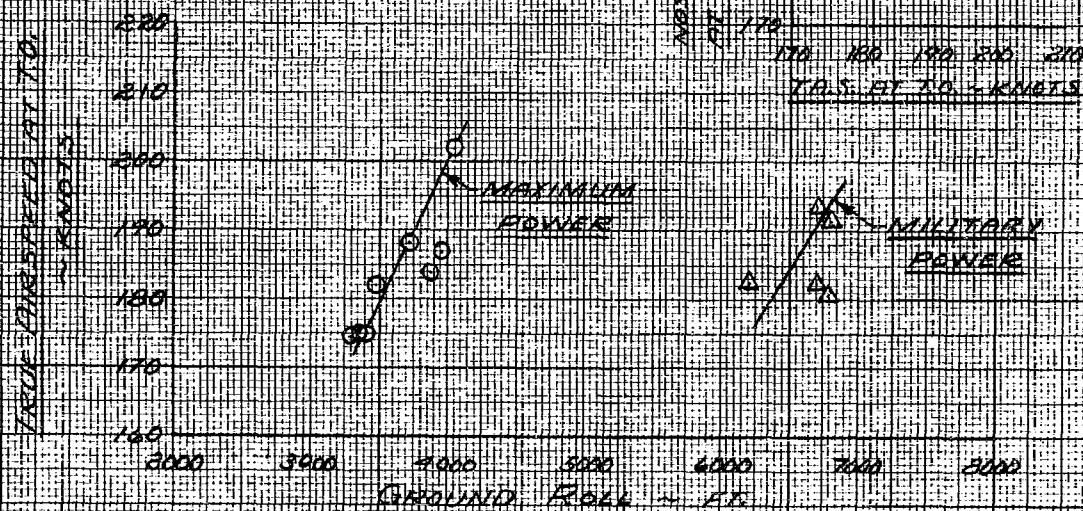


FIG. No. 2
CLIMB PERFORMANCE
F-105B USAF NO. 54-100
MAXIMUM POWER
YF-15 R-3 ENGINE

GROSS WEIGHT AT ENCL. START 27,900 LBS.

NOTE:

- (1) LEADING EDGE FLAPS 30% DOWN
 (2) 1300 POUNDS OF FUEL
REQUIRED FOR
TAKE-OFF AND ACCELERATION
TO CLIMB SPEED

- (3) DATA IS CORRECTED
TO A STANDARD
ATMOSPHERE OF 15.5%
ABOVE 35,332 FEET

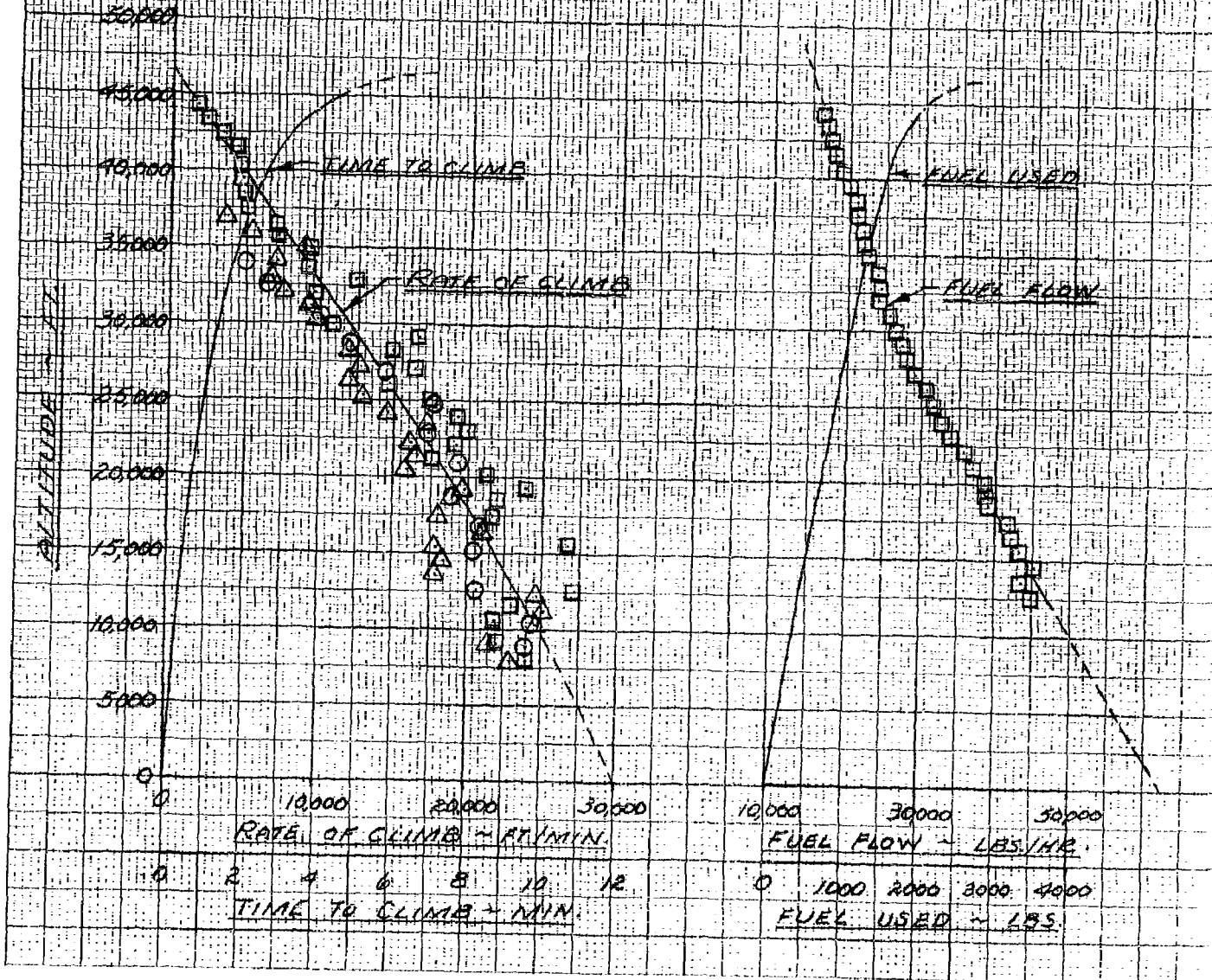


FIG. NO. 2 (CONT'D)

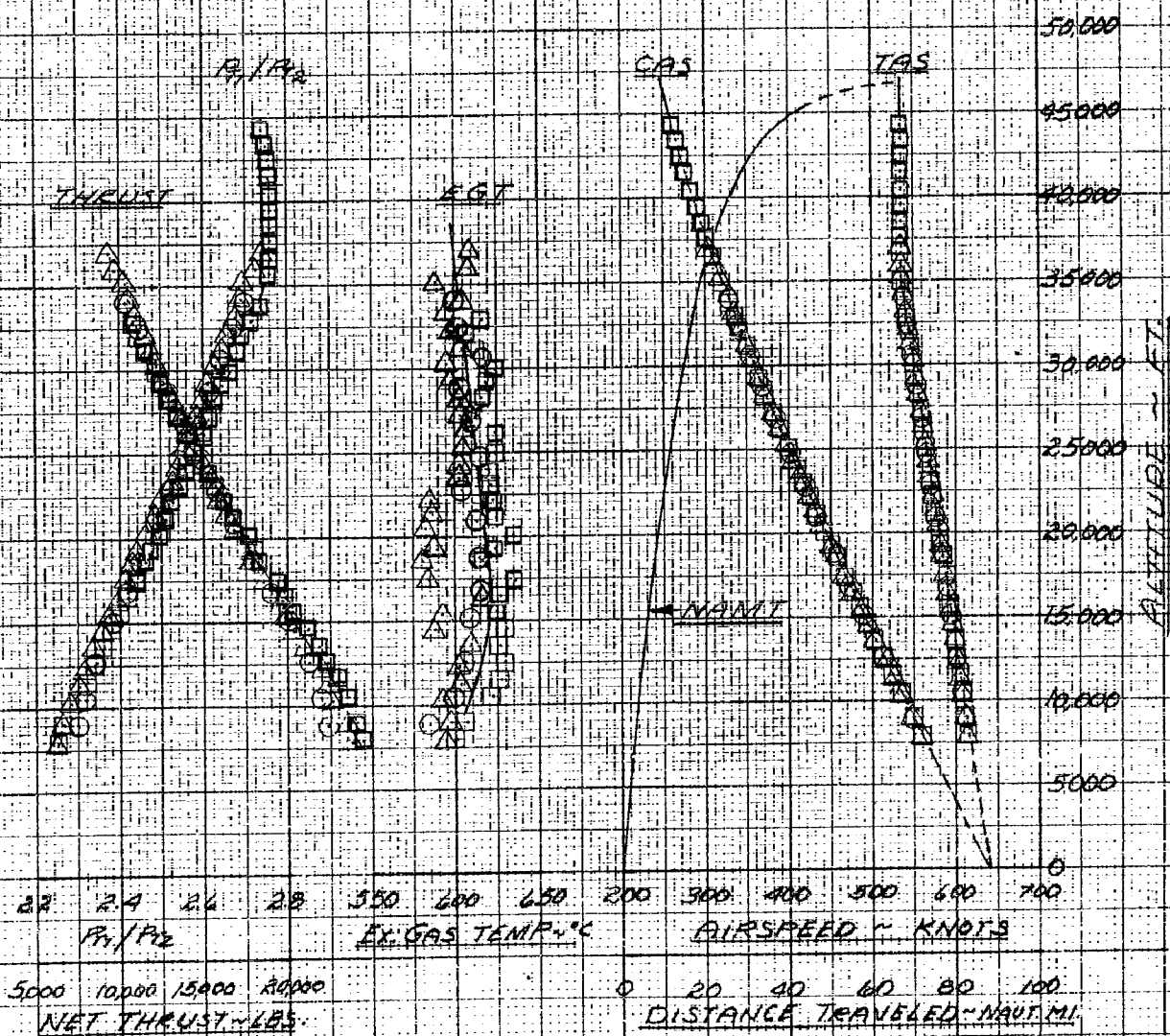


FIG. NO. 3
CLIMB PERFORMANCE
F-105B USAF NO. 54-100
YJ-75-P-3 ENGINE
MILITARY POWER

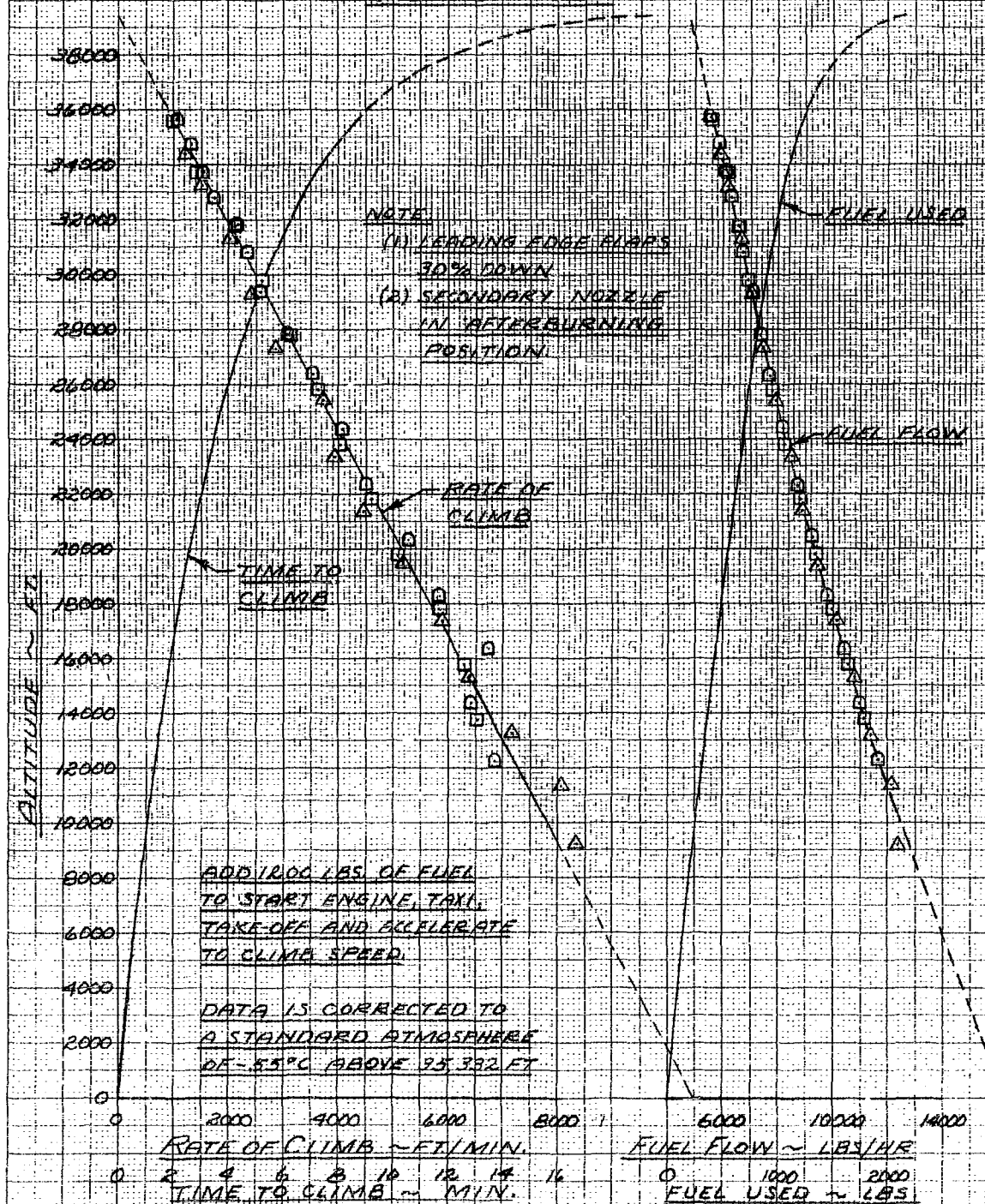


FIG. NO. 3 (CONT.)
CLIMB PERFORMANCE
F-105B USAF NO. 54-100
YJ-75-P-3 ENGINE
MILITARY POWER

GROSS WEIGHT AT ENG. START 37,900 LBS.

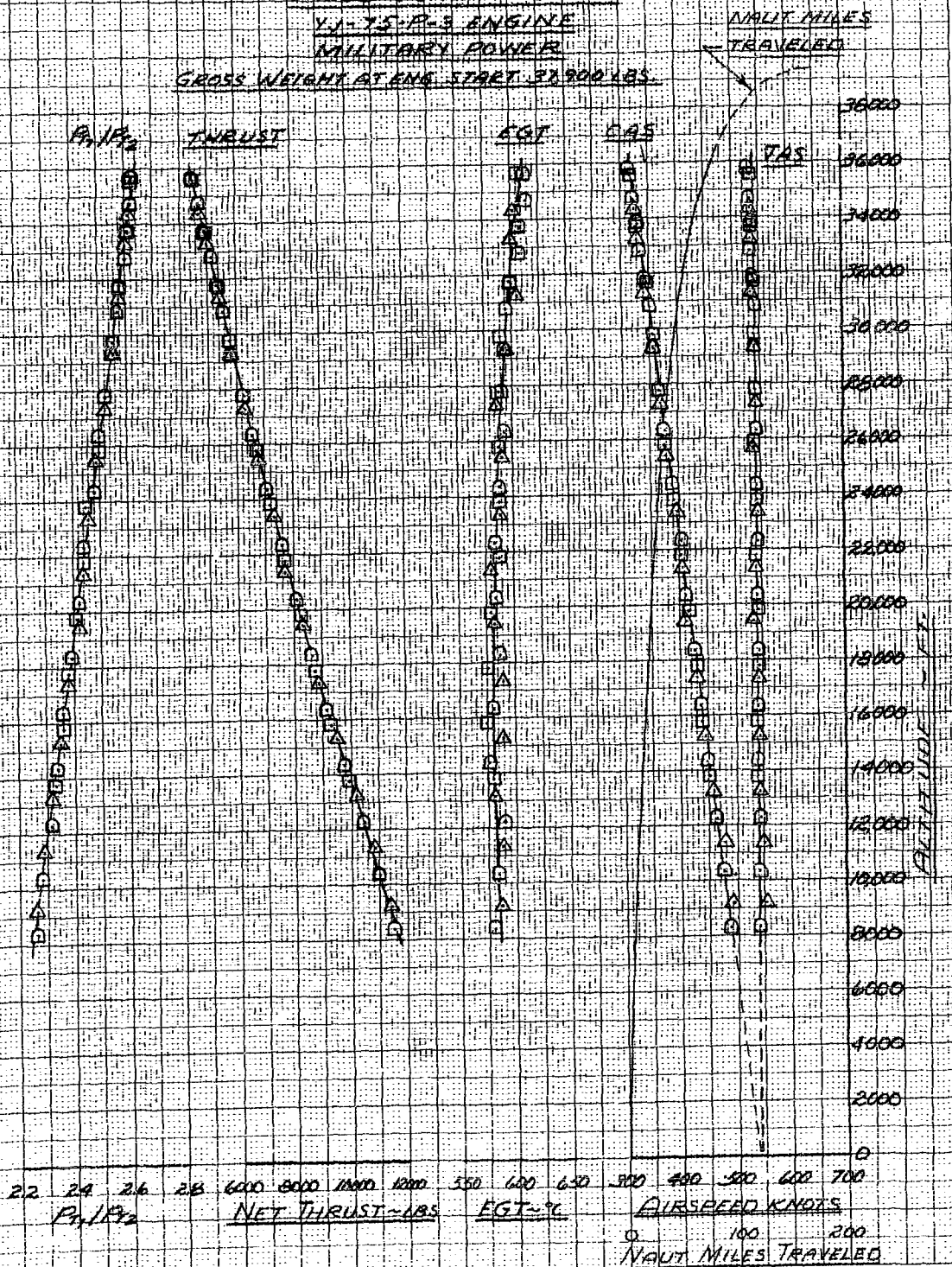


FIG. NO. 4
CLIMB POTENTIAL
F-105B, USAF NO. 54-100
YJ-75-P-3 ENGINE
BLEED DOOR SCHEDULE "C"
MAXIMUM POWER
LEADING EDGE FLAPS UP

SYM	ALT. - FT.	GR. WT.
		LBS.
△	20,000	33,000
□	35,000	38,000

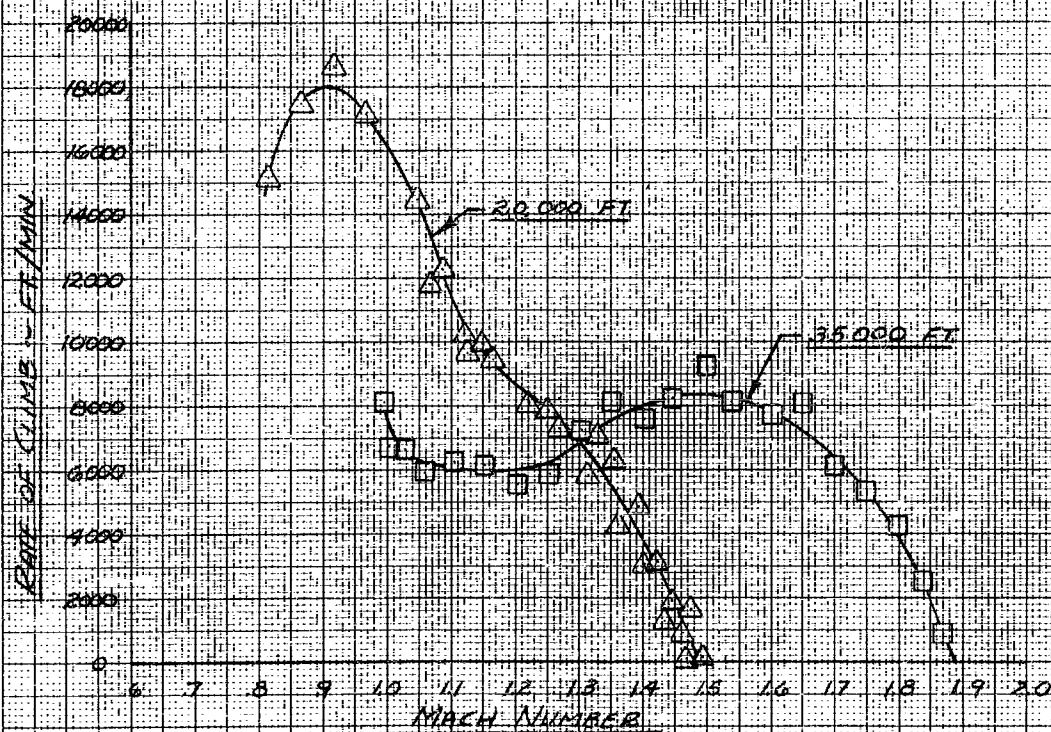


FIG. NO. 5
 CLIMB POTENTIAL
 F-105B USEF NO. 55-100
 W-75 F-3 ENGINE
 MAXIMUM POWER
 BLEED DOOR SCHEDULE "E"
 GROSS WT. 32,000 LBS.
 ALTITUDE 55,000 FT.
 LEADING EDGE PARAS. UP

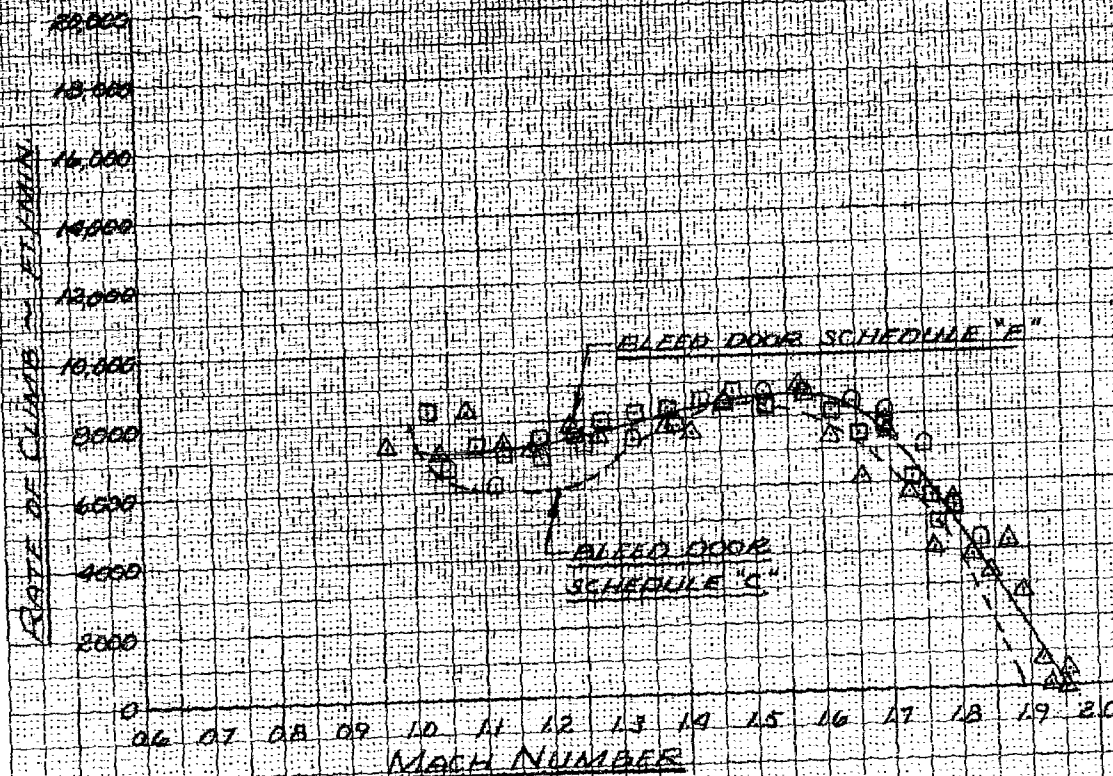


FIG. NO. 6
LEVEL ACCELERATION
F-105B USAF NO. 59-100
YJ-75-F-3 ENG. 612014
GROSS WT. 33000 LBS
ALTITUDE 20000 FT
MAXIMUM POWER
LEADING EDGE FLAPS UP
BLEED DOOR SCHEDULE C

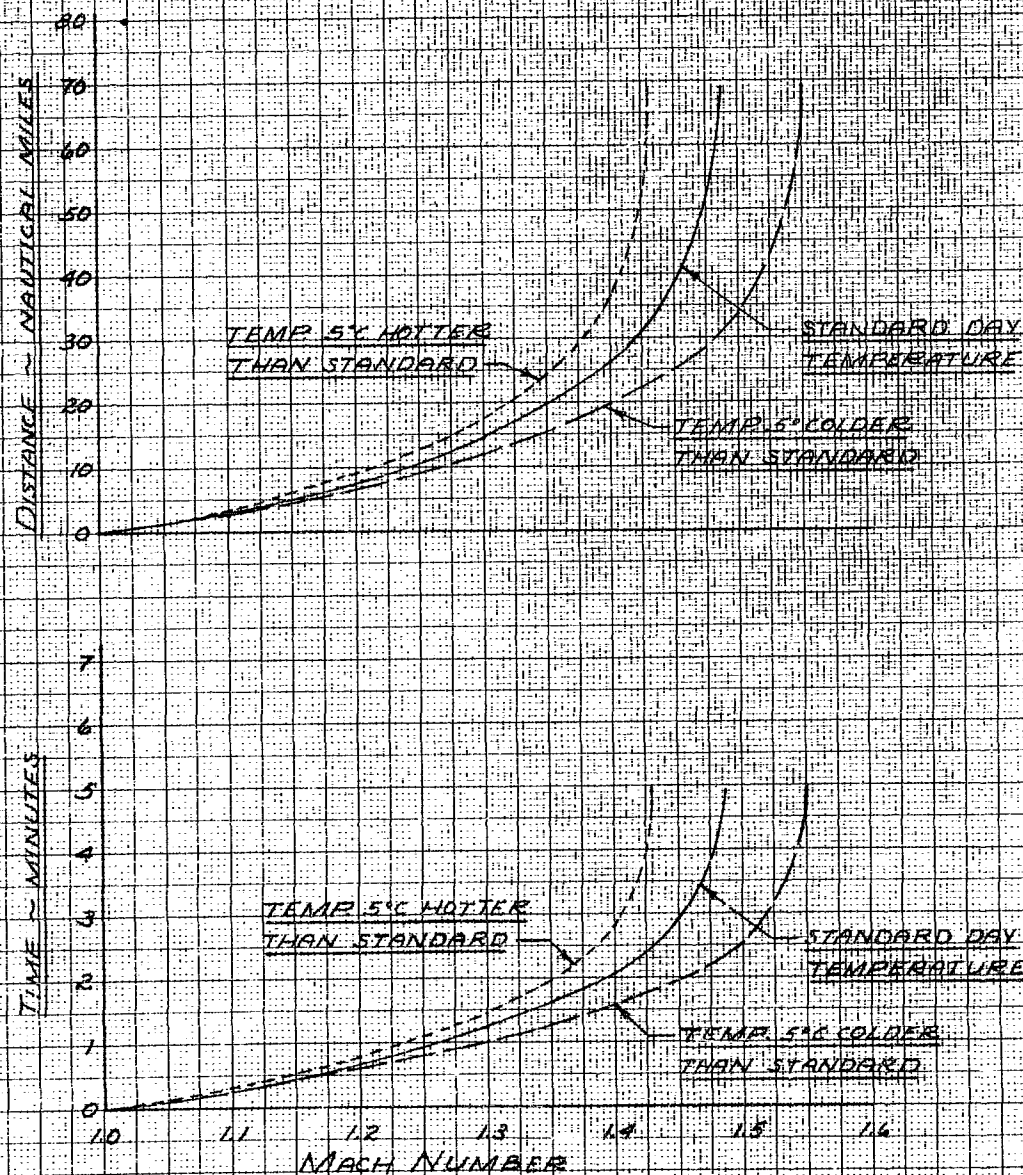


FIG. NO. 7
LEVEL ACCELERATION
F-105A USAF NO. 54-100
YJ-15-P-3 ENG. 210016
GROSS WT. 33000 LBS
ALTITUDE 20,000 FT.
MAXIMUM POWER
LEADING EDGE FLAPS UP
BLEED DOOR SCHEDULE 'C'

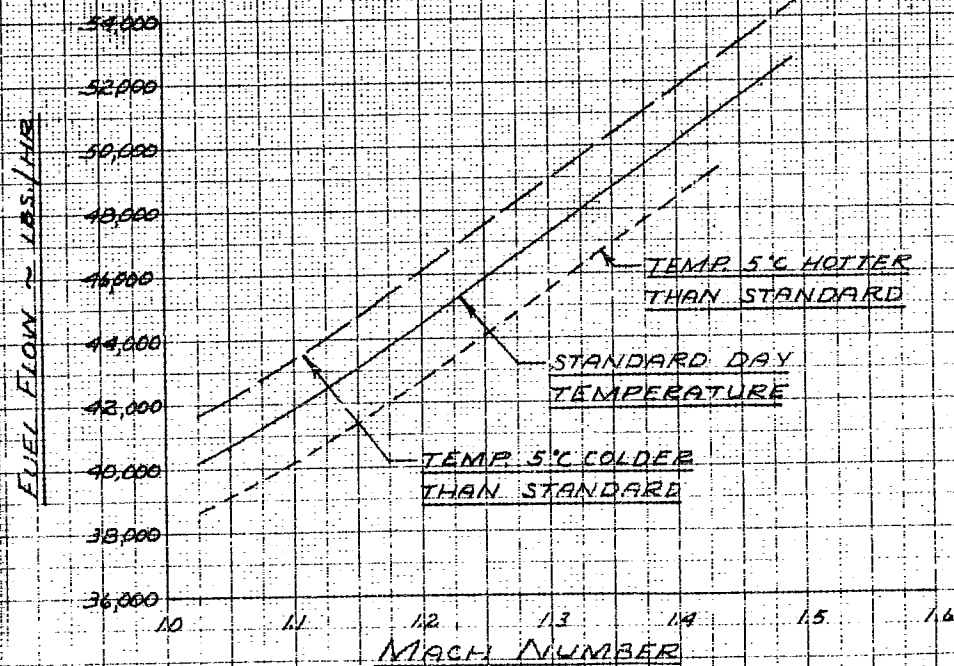
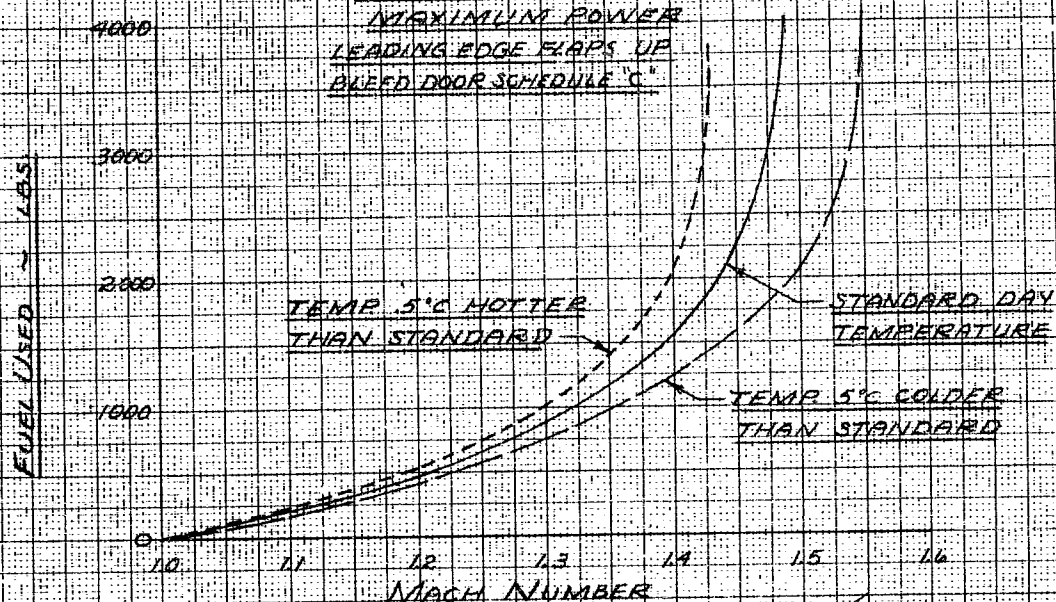


FIG No. 8
LEVEL ACCELERATION
F-105B USAF NO. 59-100
YJ-75 P-3 ENG. 610016
BLEED DOOR SCHEDULE "C"
ALTITUDE 35000 FT.
MAXIMUM POWER
GROSS WEIGHT 32000 LBS.
LEADING EDGE FLAPS UP

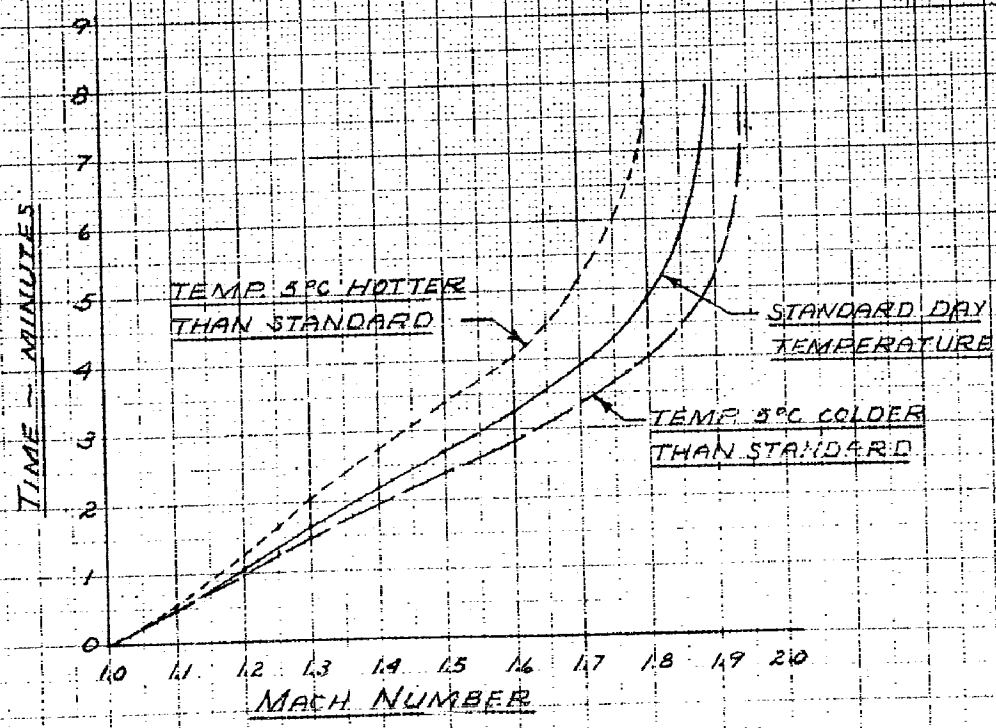
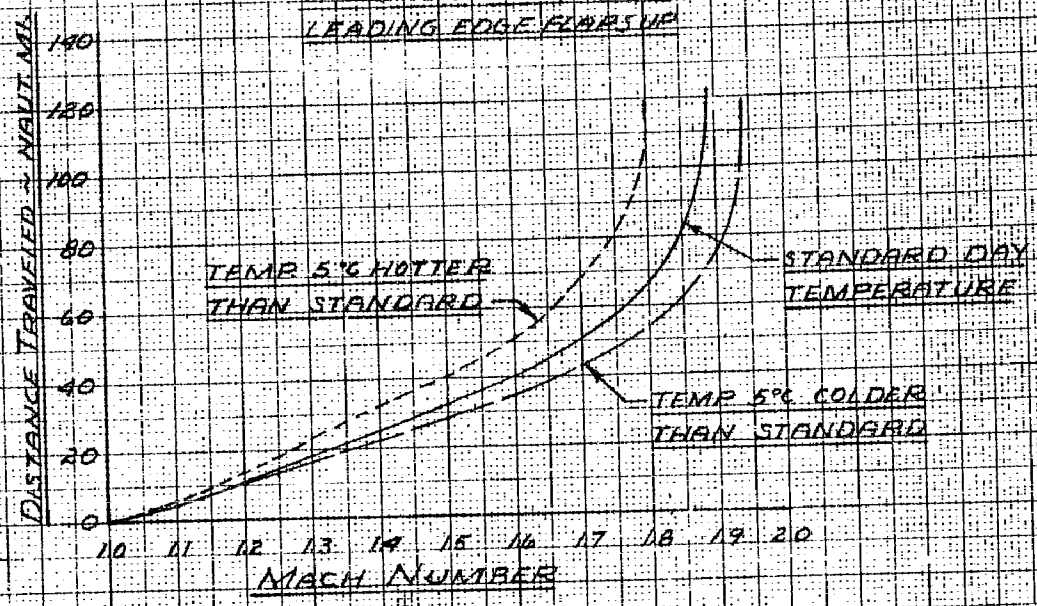


FIG. No. 9
 LEVEL ACCELERATION
 F105B USAF NO. 54-100
 YU-75 P-3 ENG. 610029
 BLEED GOOD SCHEDULE "F"
 ALTITUDE 35000 FT.
 MAXIMUM POWER
 GROSS WGT 32000 LBS.
 LEADING EDGE FLAPS UP

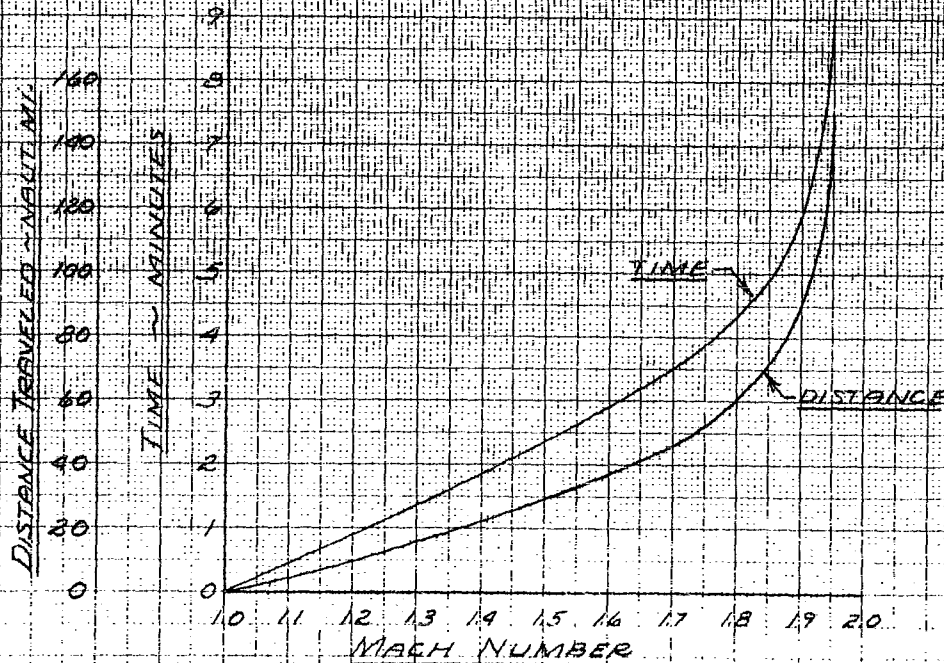
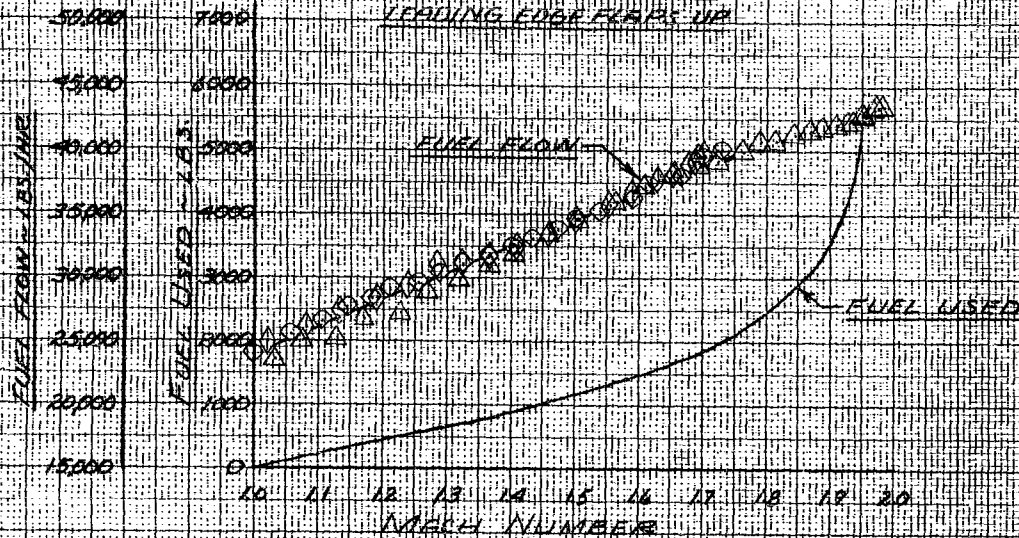


FIG. NO. 10
LEVEL FLIGHT PERFORMANCE
F-105B USAF NO. 54-100
YF-75-P-3 ENG. 610016
ALTITUDE 20,000 FEET

SYM

- STABILIZED POINTS - NON AFTERBURNING
- STABILIZED POINTS - AFTERBURNING
- ACCELERATION - MAXIMUM POWER
- △ ACCELERATION - MAXIMUM POWER

LEADING EDGE FLAPS

SUBSONIC: 30% DOWN

SUPERSONIC: UP

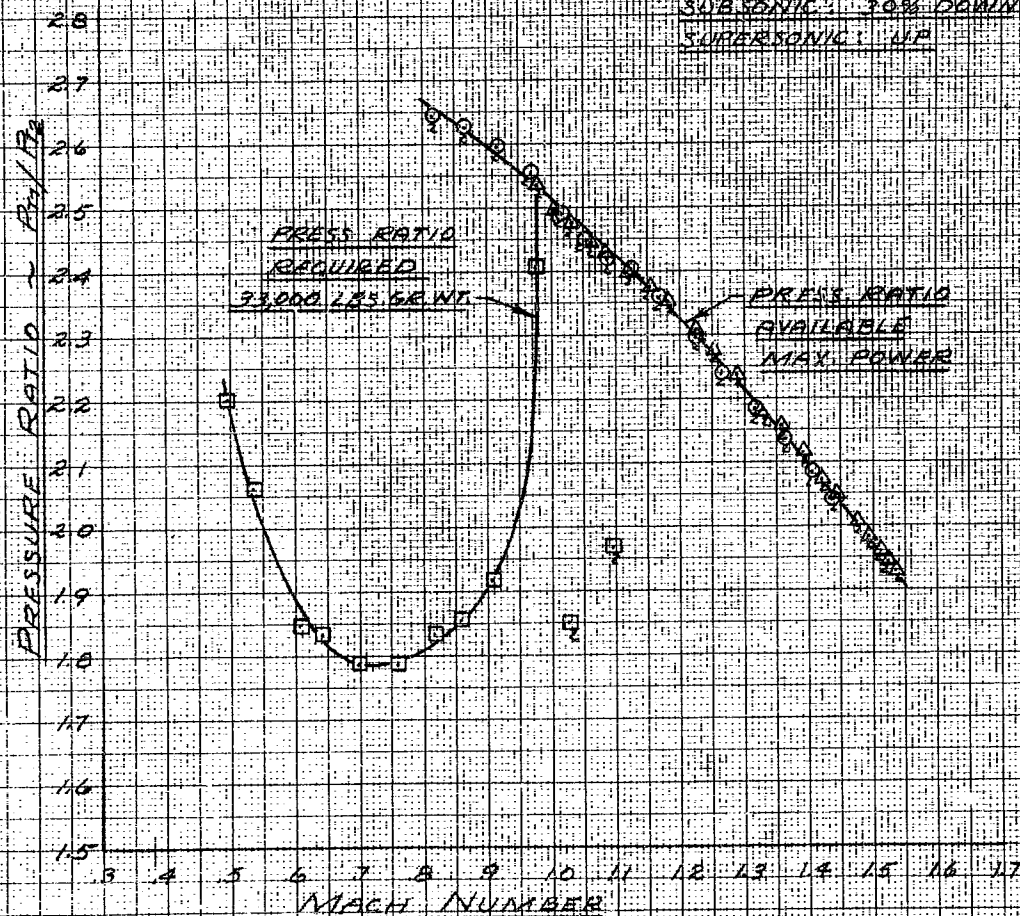


FIG. NO. 11
LEVEL FLIGHT PERFORMANCE

F-105B USAF NO. 59-100

GROSS WEIGHT 33000 LBS

ALTITUDE 20000 FEET

YJ-75-P-3 ENG. 610016

BLEED DOOR SCHEDULE C

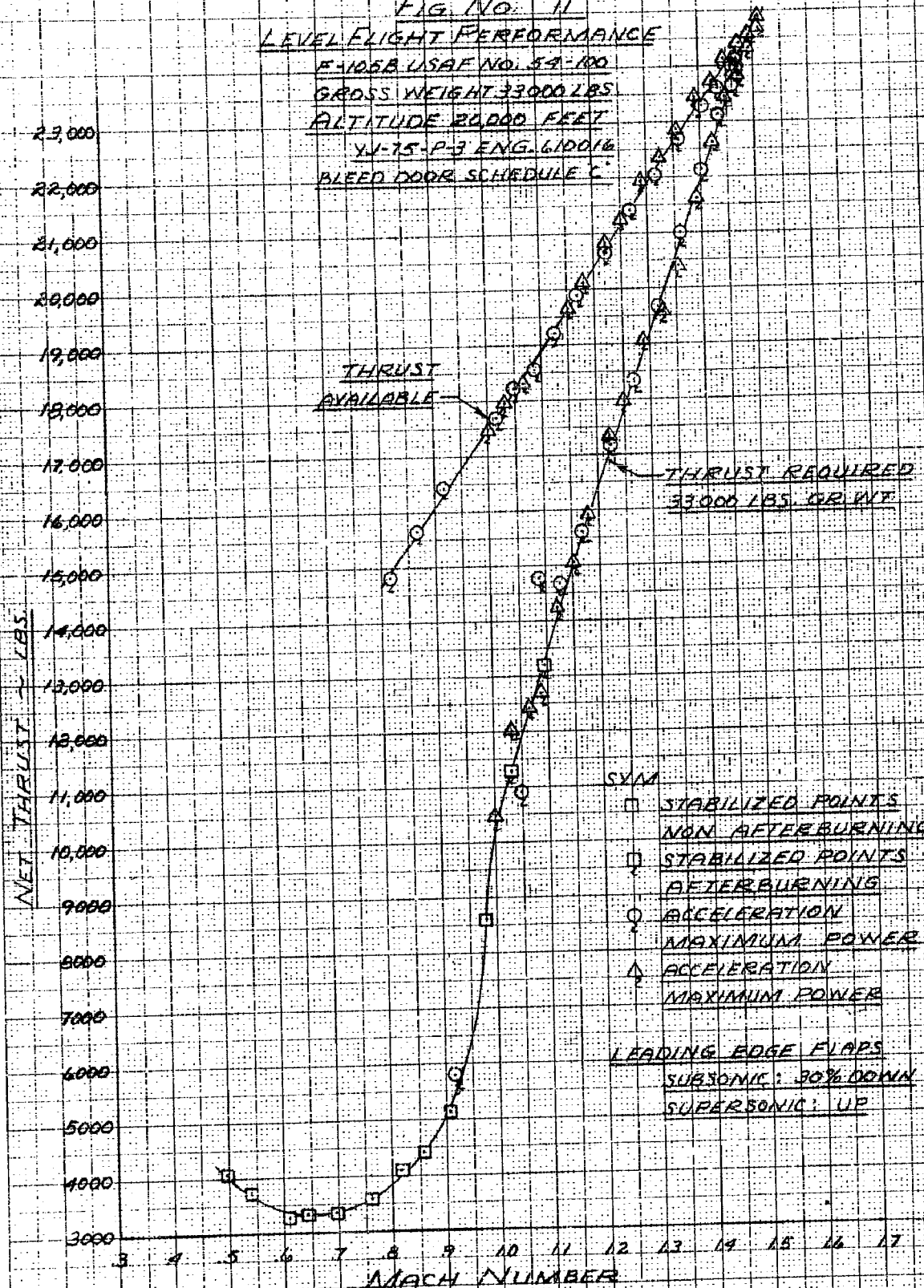


FIG. NO. 12
LEVEL FLIGHT PERFORMANCE
F-105B USAF NO. 54-100
YJ-75-P-3 ENG 610016
BLEED DOOR SCHEDULE C
ALTITUDE 35,000 FT

SYM

- STABILIZED POINTS - NON AFTERBURNING
- STABILIZED POINT - AFTERBURNING
- ACCELERATION - MAXIMUM POWER

LEADING EDGE FLAPS

SUBSONIC: 30% DOWN

SUPERSONIC: UP

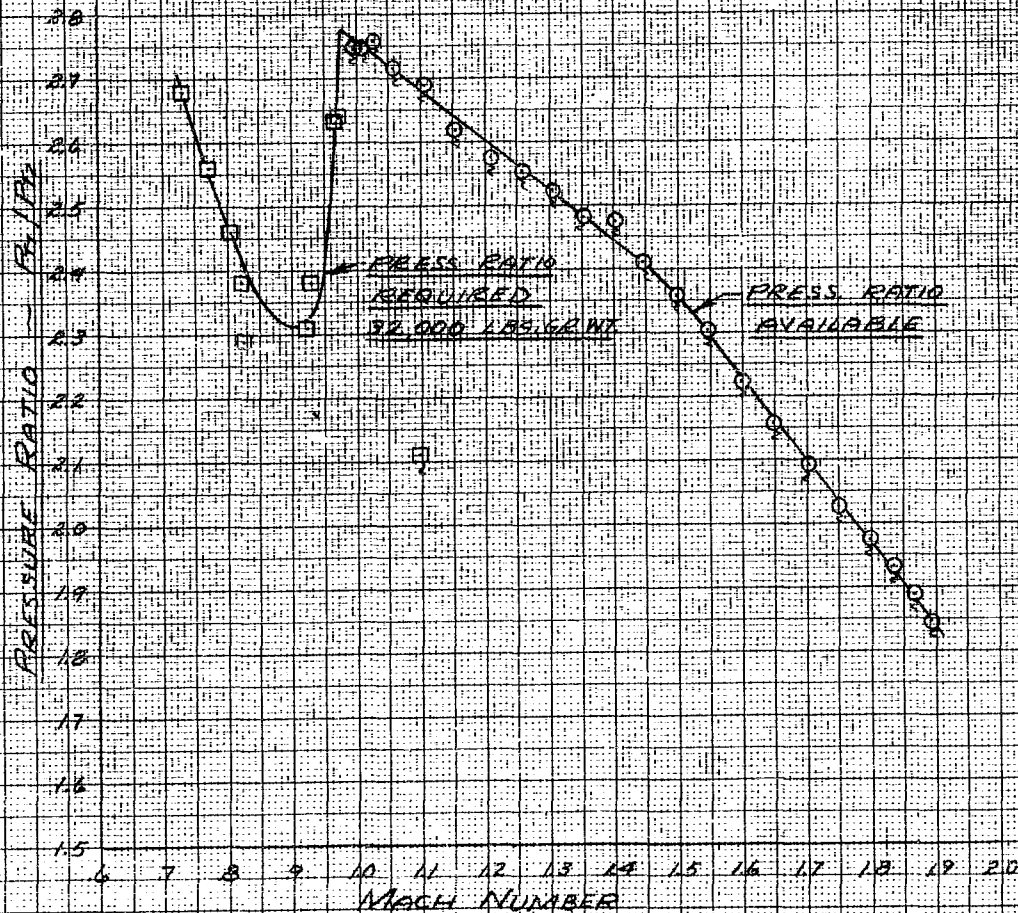


FIG. NO. 13
LEVEL FLIGHT PERFORMANCE
F-105B USAF NO. 54-100
YT-75-P-3 ENG. 610016
FIXED DOOR SCHEDULE C
ALTITUDE 35000 FT
GROSS WT 32000 LBS

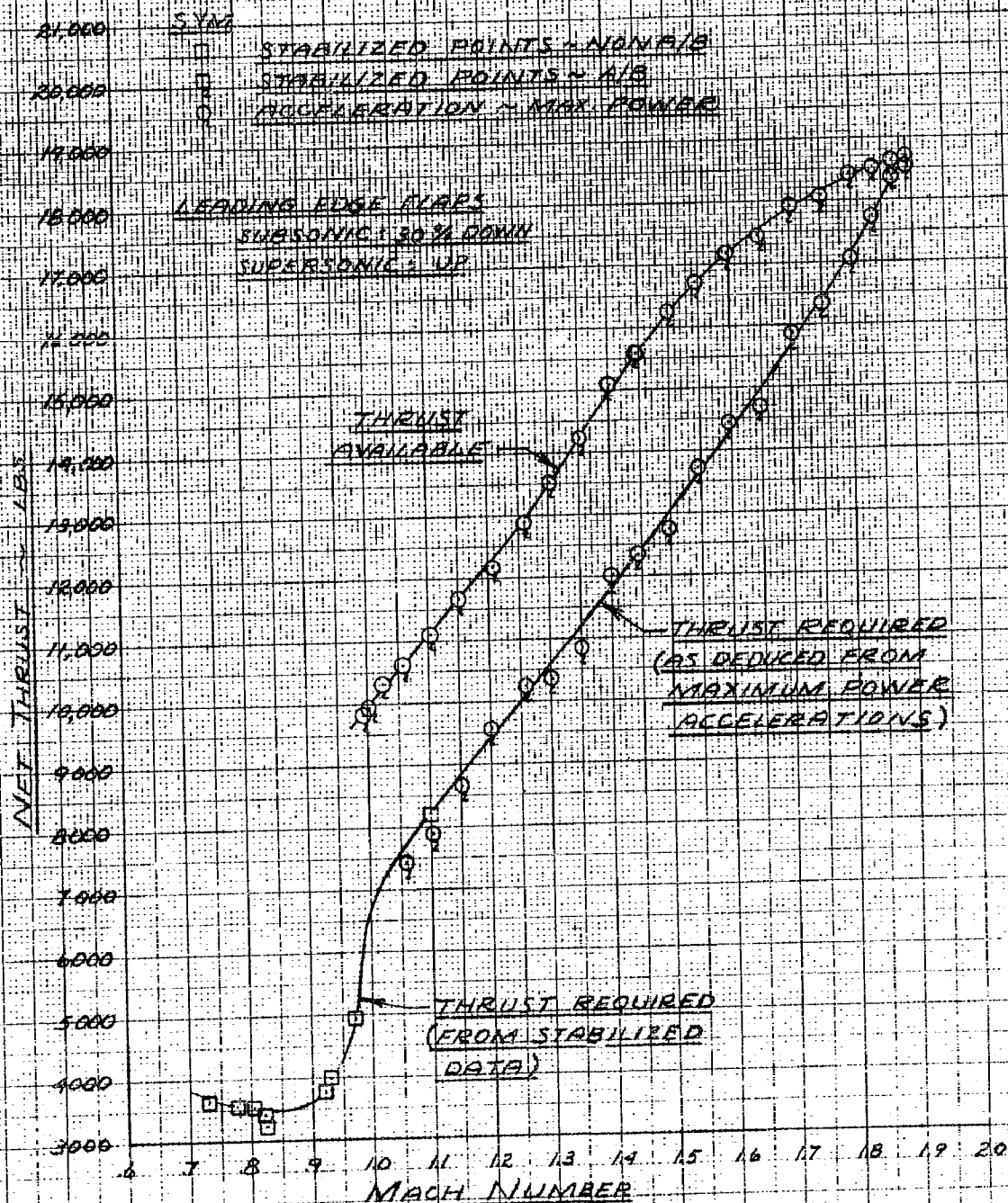


FIG. NO. 14
LEVEL FLIGHT PERFORMANCE
F-105B USAF NO. 54-100
YJ-75-P-3 ENG. 610023
BLEED DOOR SCHEDULE "F"
ALTITUDE 35000 FT
GROSS WT. 32000 LBS.

LEADING EDGE FLAPS
SUBSONIC: 30% DOWN
SUPERSONIC: UP

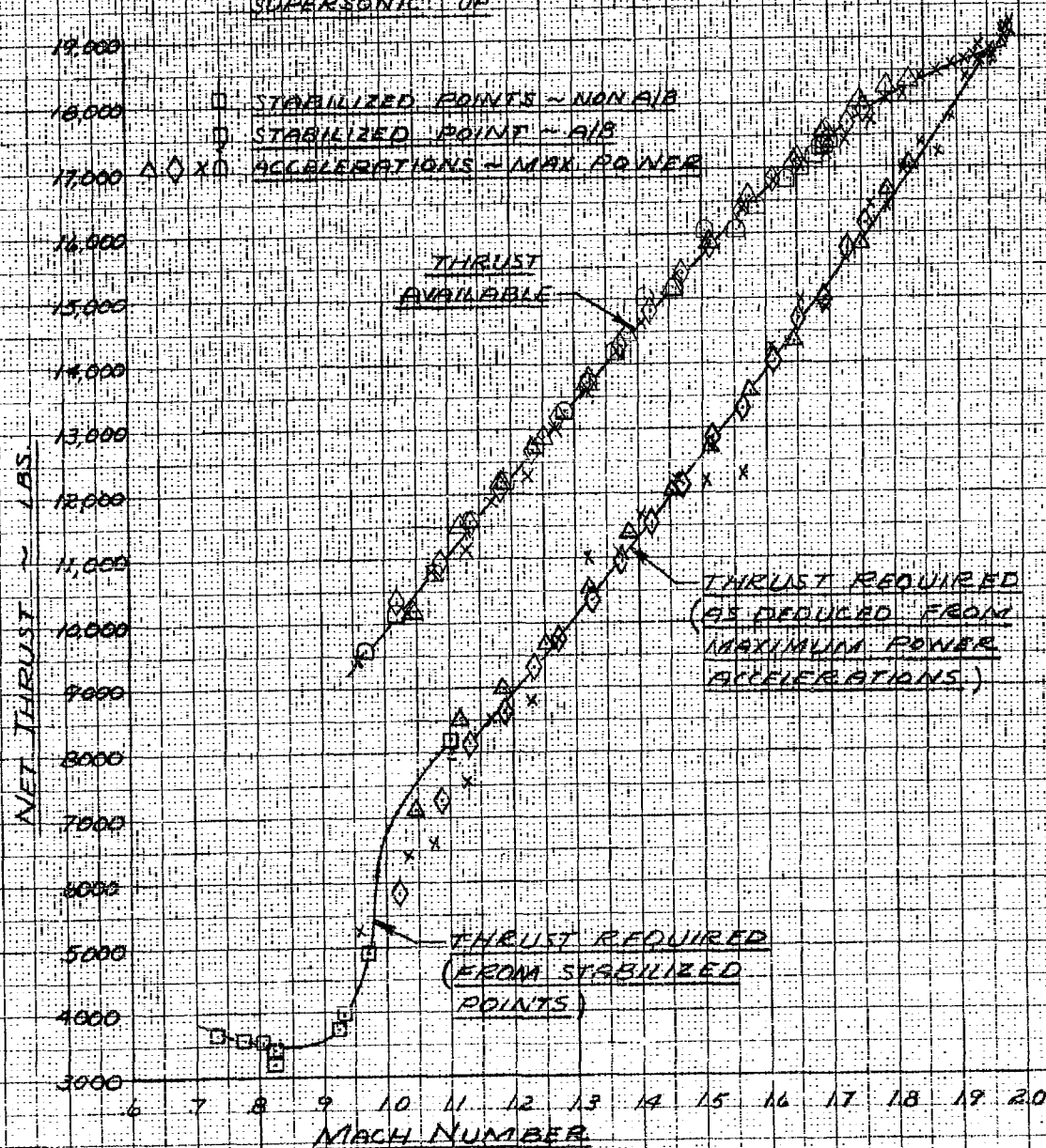


FIG. NO. 15
NAUTICAL AIR MILES PER POUND OF FUEL
F-105B USAF NO. 54-100
NO EXTERNAL STORES
YJ-75-P-3 ENGINE 610019

<u>SYM</u>	<u>ALT-FT</u>	<u>GR. WT. -LBS.</u>	
■	20,000	33,000	<u>NON AFTERBURNING</u>
■	20,000	33,000	<u>AFTERBURNING</u>
□	35,000	32,000	<u>NON AFTERBURNING</u>

LEADING EDGE FLAPS
SUBSONIC: 30% DOWN
SUPERSONIC: UP

SECONDARY NOZZLE IN
AFTERBURNING POSITION

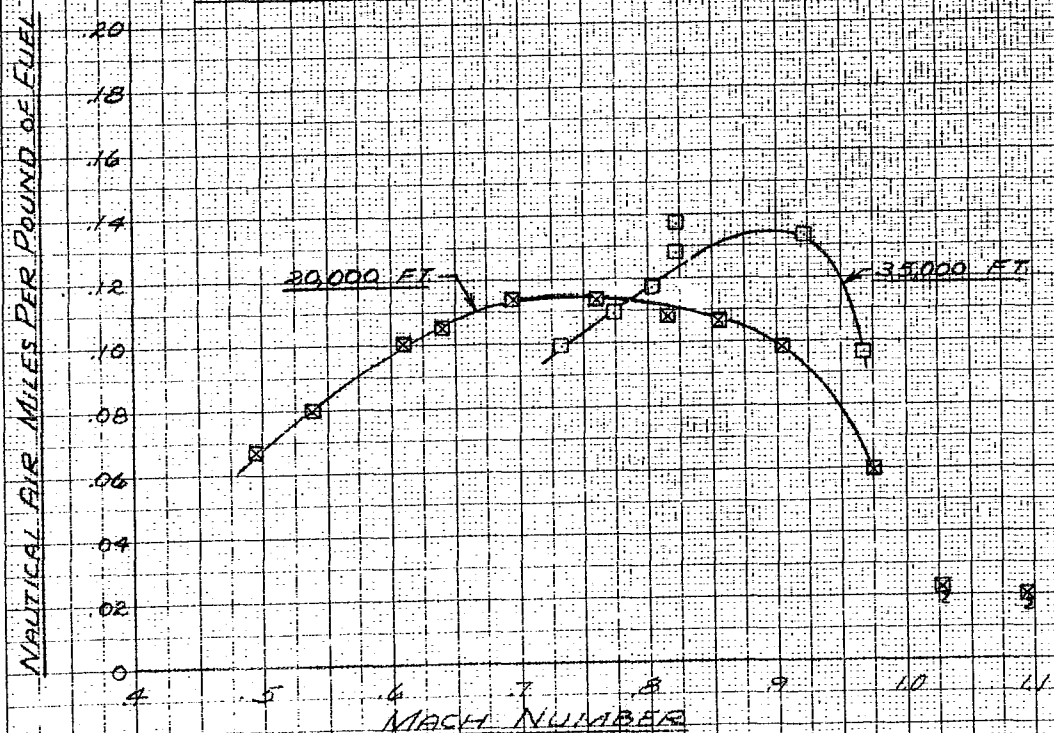


FIG. NO. 16
DRAG POLAR
F-105R USE NO. 59-100
YJ-75-P-3 ENGINE
CRUISE CONFIGURATION
NO EXTERNAL STORES

LEADING EDGE FLAPS
SUBSONIC: 30% DOWN
SUPERSONIC: UP

	ALT.	GR. WT.
SYM.	~FT.	~LBS.
△	20,000	33,000
○	35,000	32,000

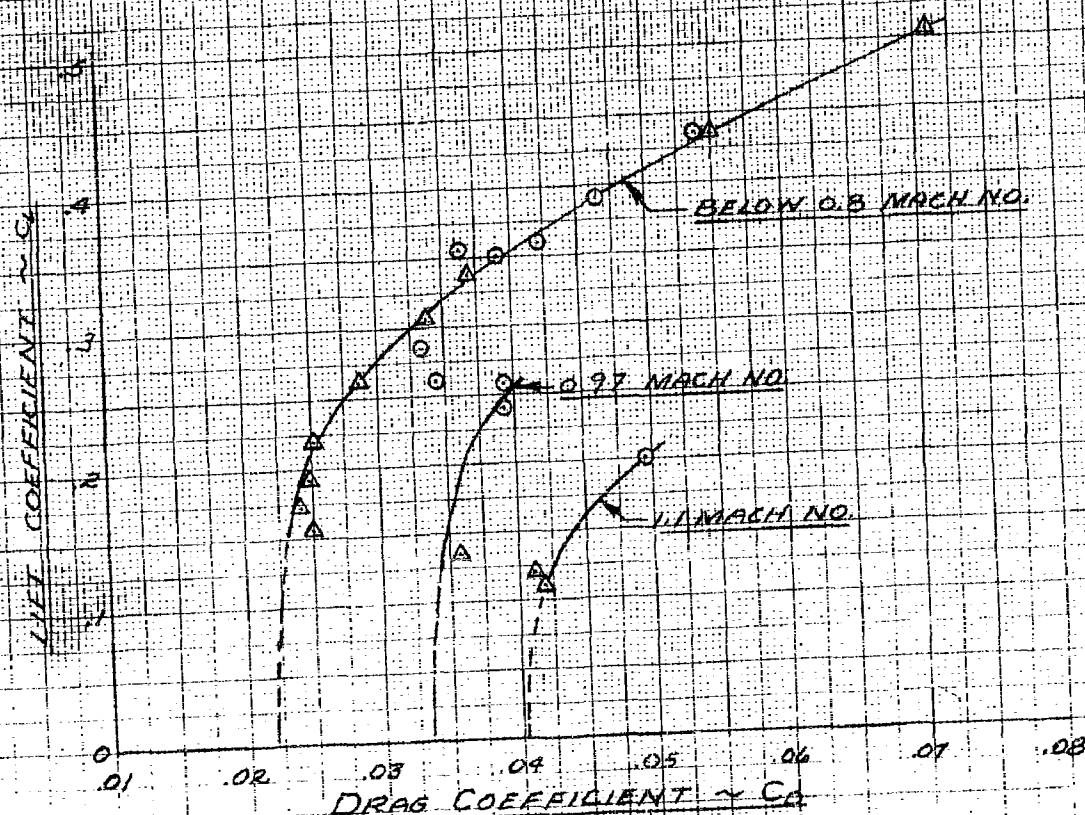


FIG. NO. 17
MANEUVERING PERFORMANCE
F-105B USAF NO. 54-100
YU-28-P-3 ENG. 618023
MAXIMUM POWER
GROSS WT. 32,000 LBS.
ALTITUDE 35,000 FEET
LEADING EDGE FLAPS UP
BLEED DOOR SCHEDULE 'F'

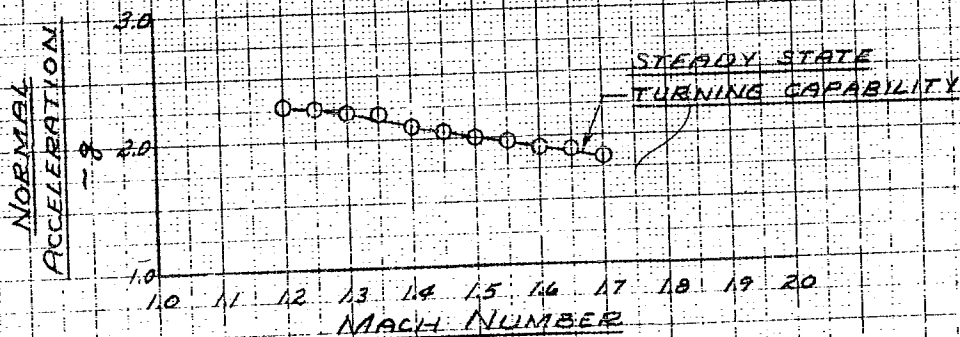
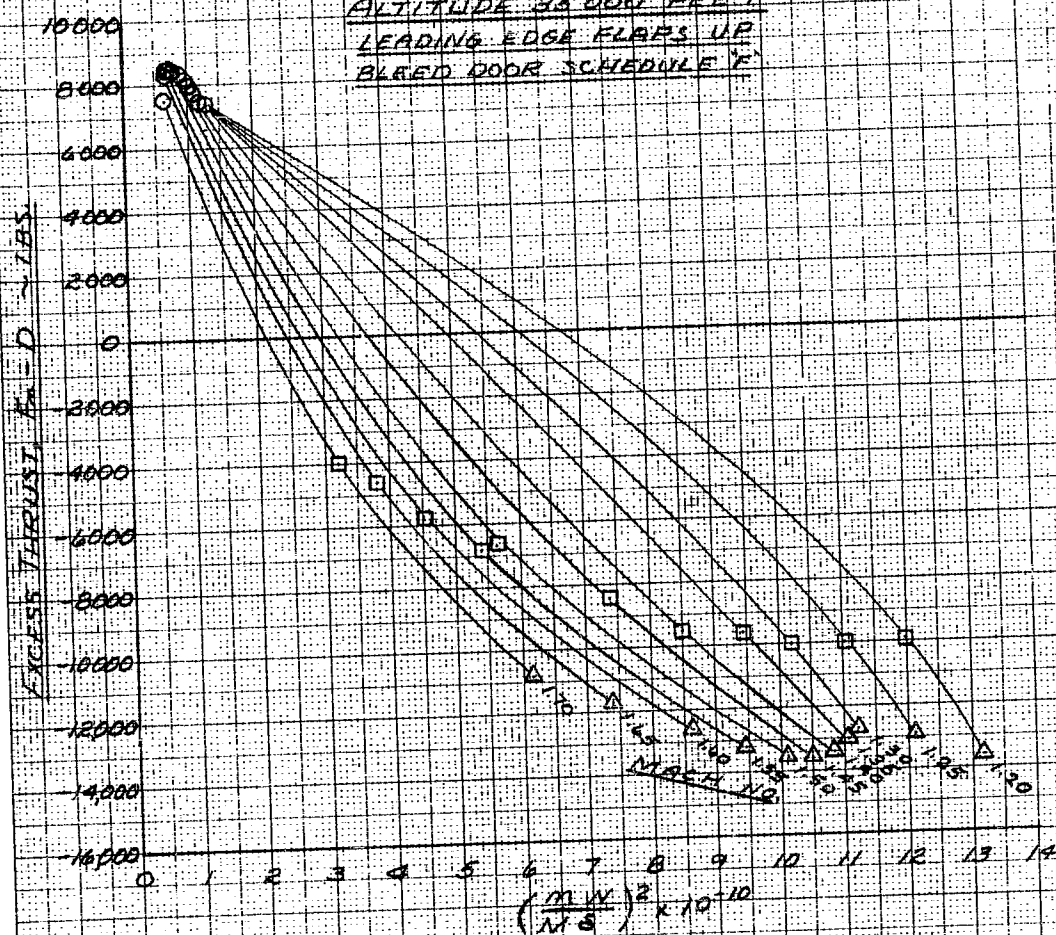


FIG NO. 18
 ZOOM CLIMB
 F-105B USAF NO. 59-100
 YN-75-R-3 ENG 610023
 MAXIMUM POWER
 BLEED DOOR SCHEDULE "F"
 LEADING EDGE FLAPS UP

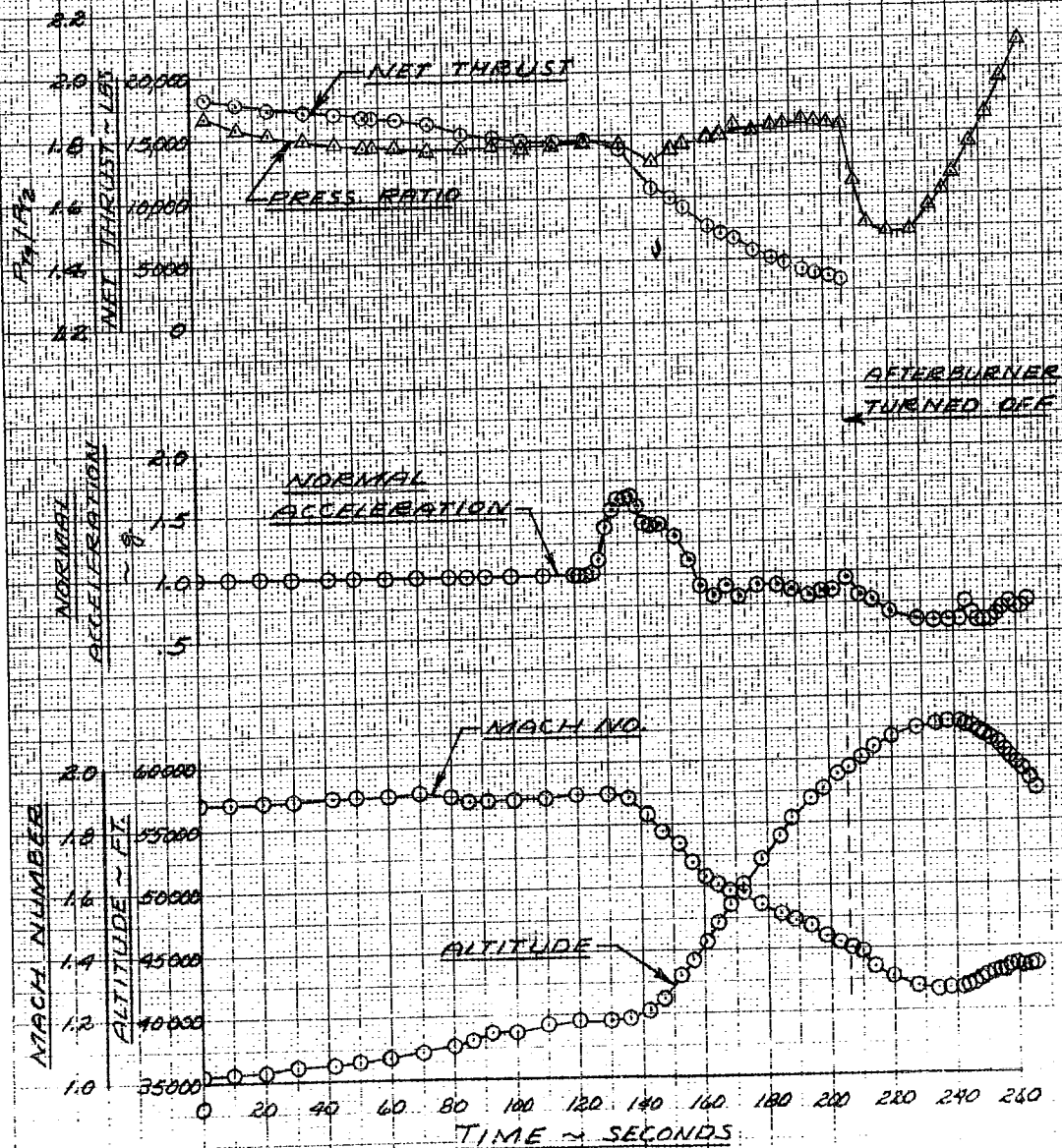


FIG. NO. 19
 CORRECTED FUEL FLOW
 F-105B USAF NO. 59-100
 YJ-75-P-3 ENG. 410016
 NON-AFTERBURNING

SYM	ALT-FT	
□	20000	SPEED-POWER
⊗	35000	SPEED-POWER

TEST FUEL FLOW CORRECTED TO INLET CONDITIONS
 W₁ 15.18, V₂ 1.05/HR

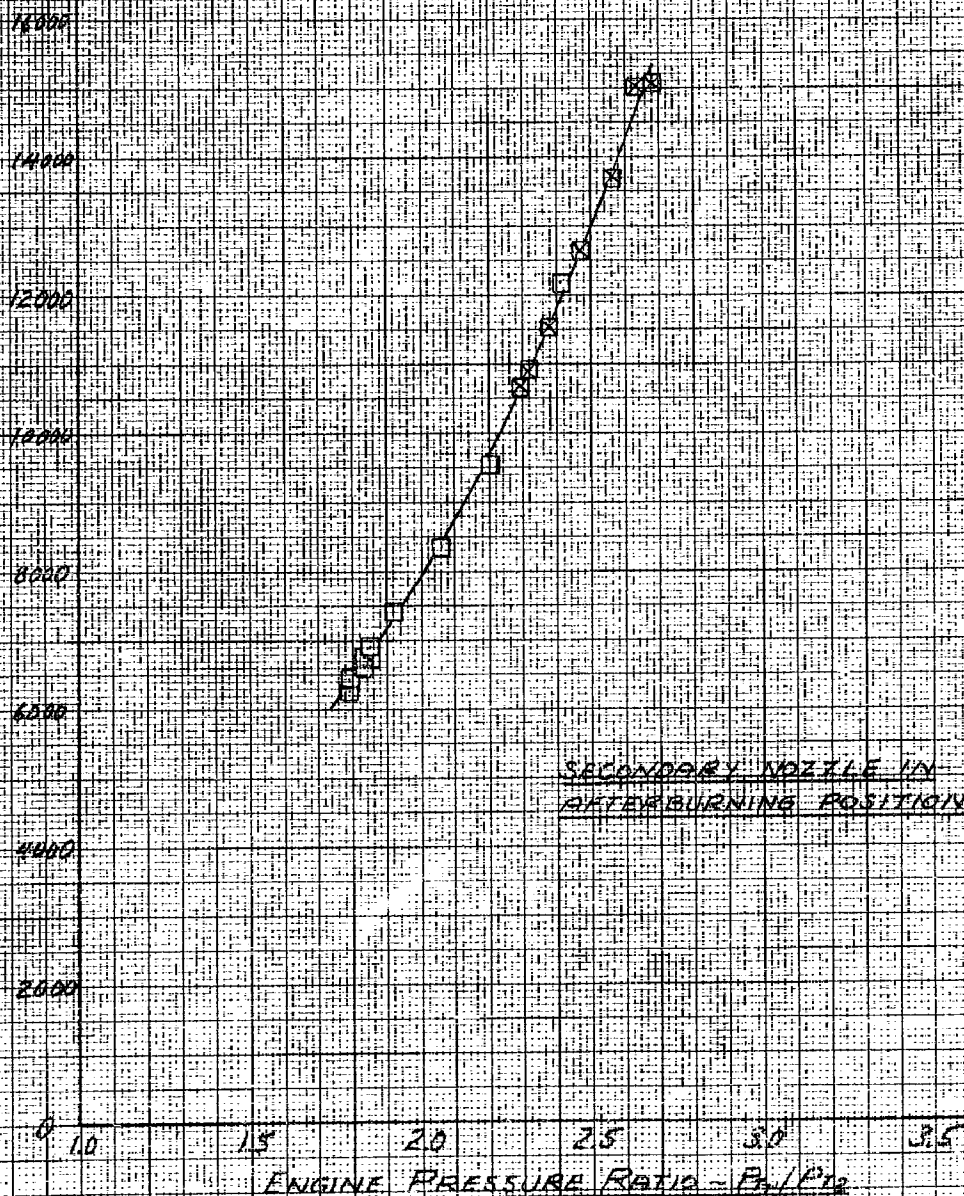


FIG. NO. 29
CORRECTED FUEL FLOW
F-105B USAF NO. 54-100
YJ-75-P-3 ENGINE SN 610016
AFTERBURNING
BLEED DOOR SCHEDULE "C"

<u>SYM.</u>	<u>ALT-FT</u>	<u>SPEED-POWER</u>
<u>□</u>	<u>20,000</u>	<u>CLIMB</u>
<u>○</u>	<u>30,000</u>	<u>ACCELERATION</u>
<u>+</u>	<u>40,000</u>	<u>ACCELERATION</u>
<u>x</u>	<u>45,000</u>	<u>ACCELERATION</u>

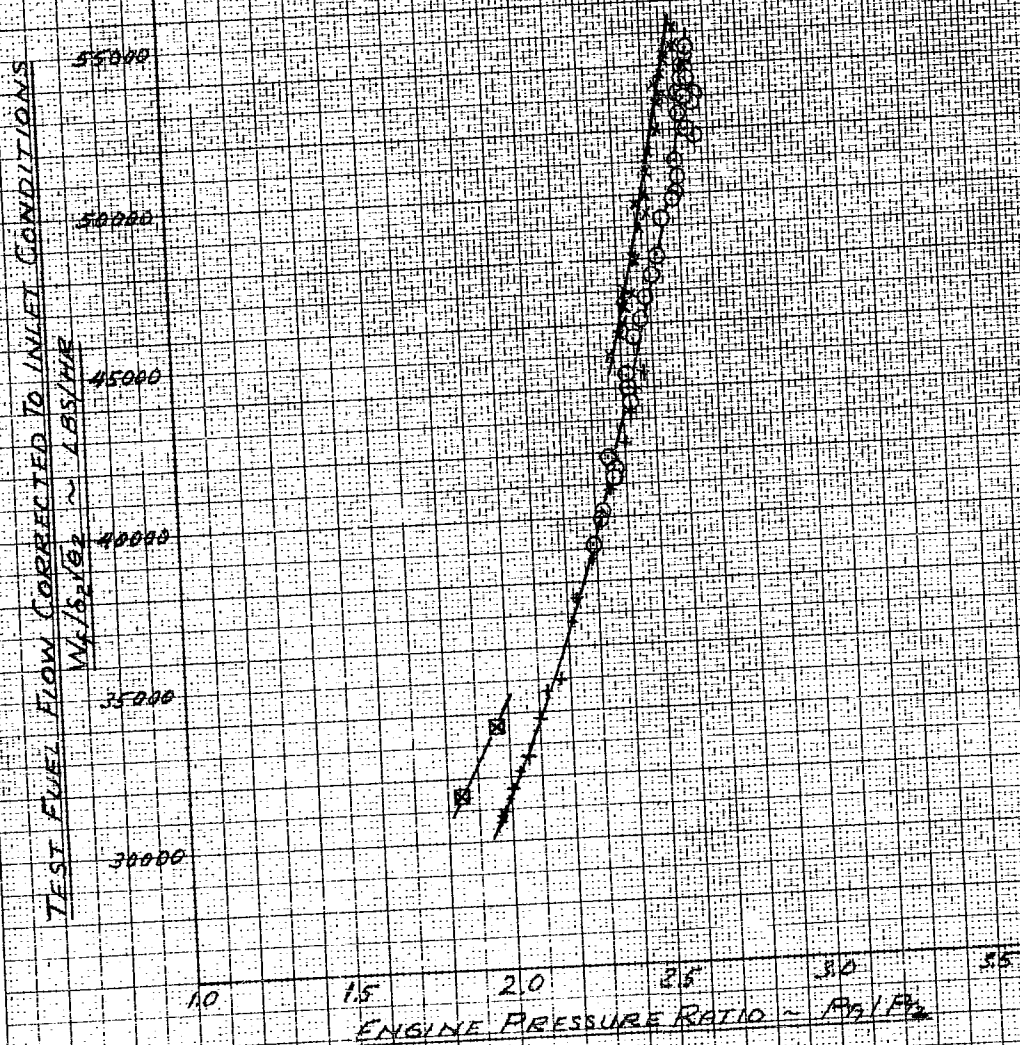


FIG. NO. 21
 CORRECTED FUEL FLOW
 F-105B USAF NO. 54-100
 YV-75-P-3 ENGINE SN 610023
 BLEED DOOR SCHEDULE "A"
 MAXIMUM POWER
 ALTITUDE 35,000 FEET

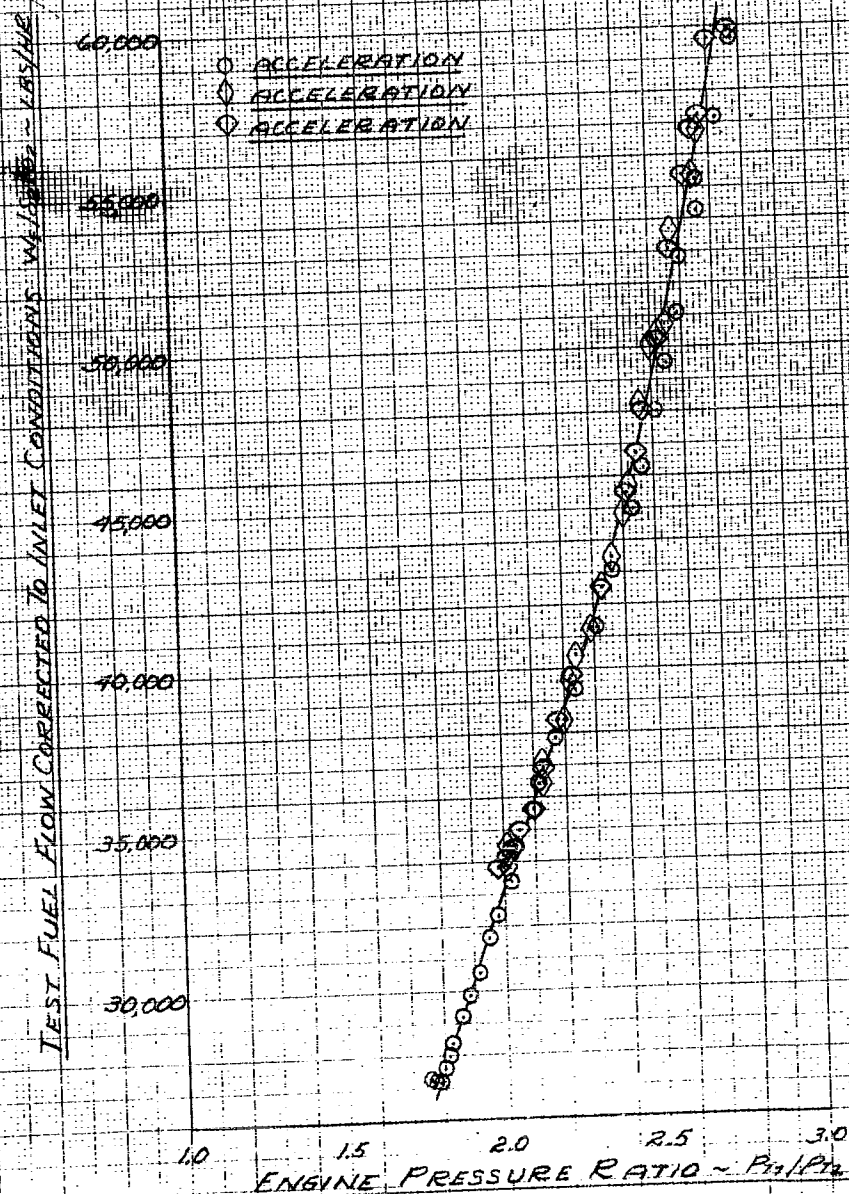


FIG NO. 22
AIR SPEED CALIBRATION
NOSE BOOM TEST SYSTEM
F-105B USEF NO. 54-100
NO EXTERNAL STORES

SYM	ALT -FT-	METHOD
□	20,000	STABILIZED FACE POINTS
△	35,000	STABILIZED FACE POINTS
○	35,000	ACCELERATION ALONG PILOT'S SMOKE TRAIL

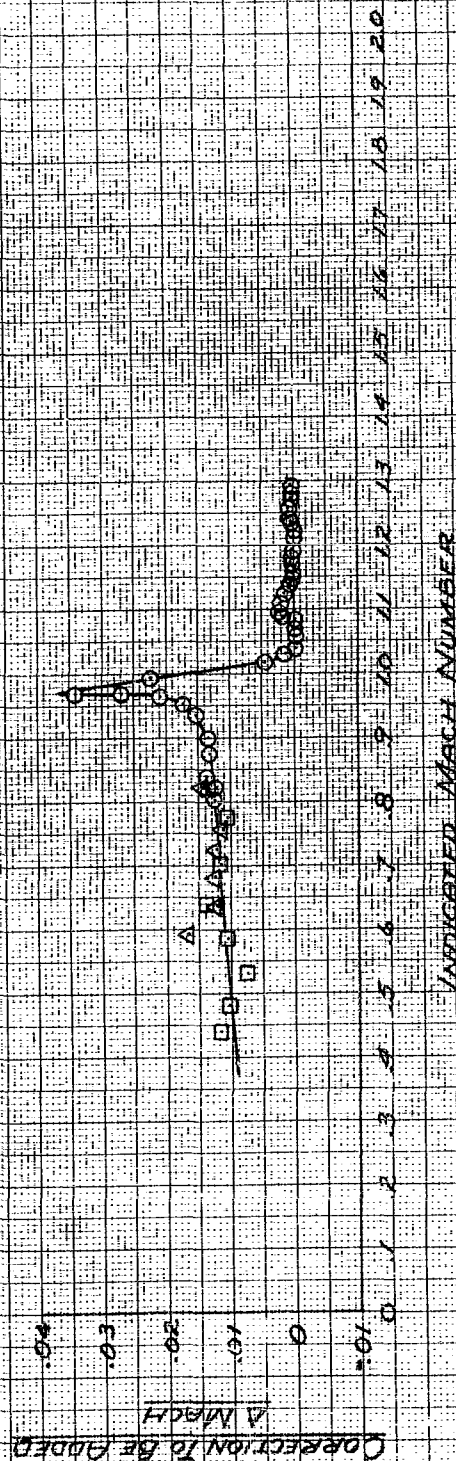
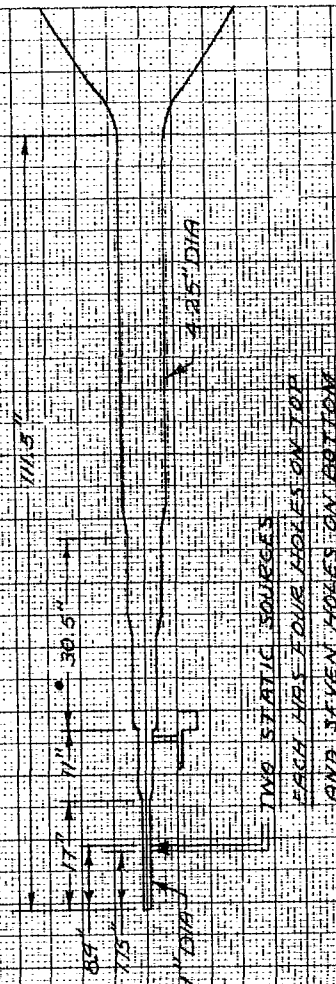


FIG NO. 23

UNSTEADY CALIBRATION
WING BOOM STANDARD SYSTEM
F-105B USAF NO. 54-100
NO EXTERNAL STORES

SYM	ALT. -FT.	METHOD
□	20,000	STABILIZED PACE POINTS
△	35,000	STABILIZED PACE POINTS
○	35,000	ACCELERATION ALONG PIPER'S SMOKE TRAIL
▽	35,000	ACCELERATION, NOSE BOOM AS REFERENCE
◇	35,000	ACCELERATION, NOSE BOOM AS REFERENCE
○	35,000	ACCELERATION, NOSE BOOM AS REFERENCE
x	20,000	ACCELERATION, NOSE BOOM AS REFERENCE

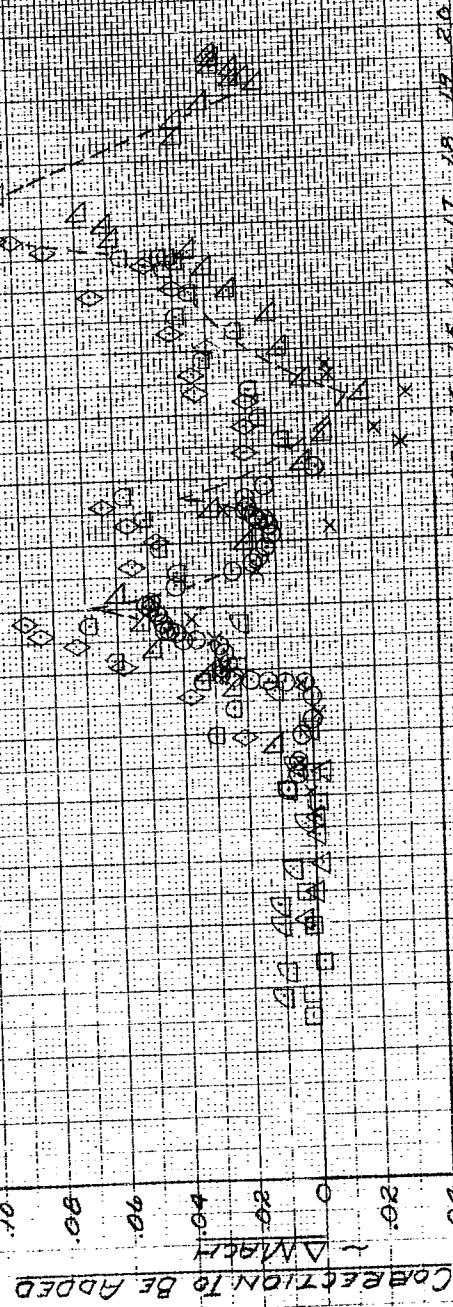
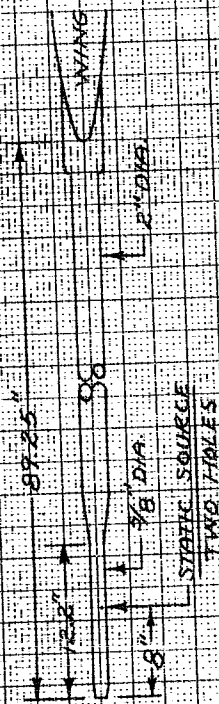


FIG. NO. 24
 PRESSURE RATIO BIAS CURVE
 F-105B USAF NO. 54-100
 YJ-75-P-3 ENGINE
 MAXIMUM POWER

- ACCELERATION 35,000 FT.
- ▽ ACCELERATION 35,000 FT.
- ACCELERATION 20,000 FT.
- ACCELERATION 20,000 FT.
- x ACCELERATION 45,000 FT.
- CLIMB
- CLIMB

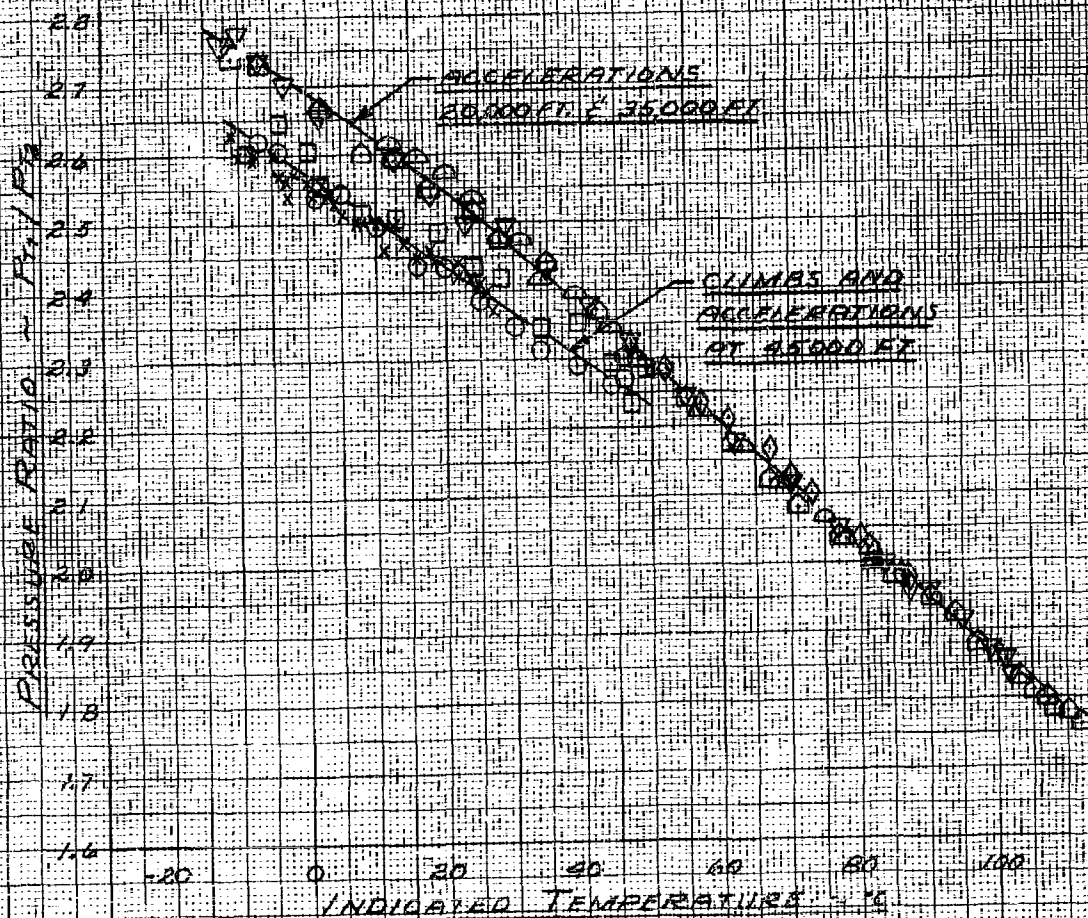
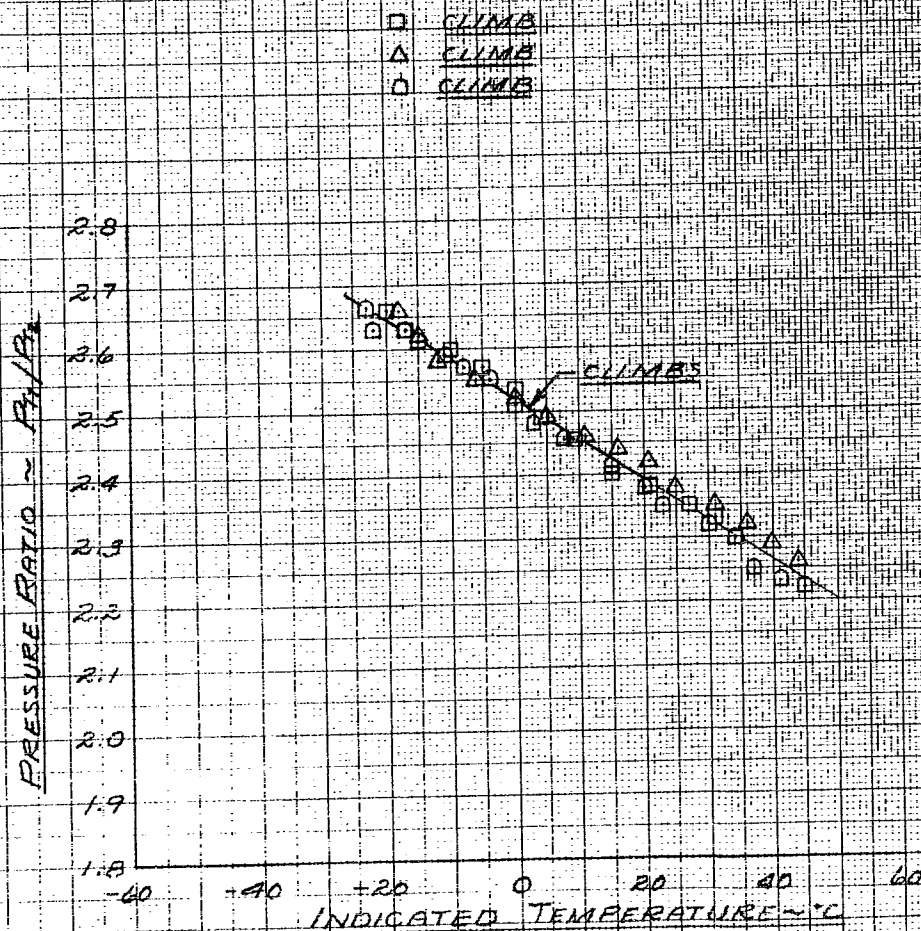


FIG NO. 25
PRESSURE RATIO BIAS CURVE
F-105B USAF NO. 54-100
YJ-75-F-3 ENGINE
MILITARY POWER



YJ-75-P-3 ENGINE

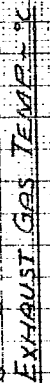


FIG NO. 27
 STATIC THRUST CALIBRATION
 F-105B USAF NO. 54-100
 ENGINE W-75-P-3 S/N 6100218
 18 FEB 57

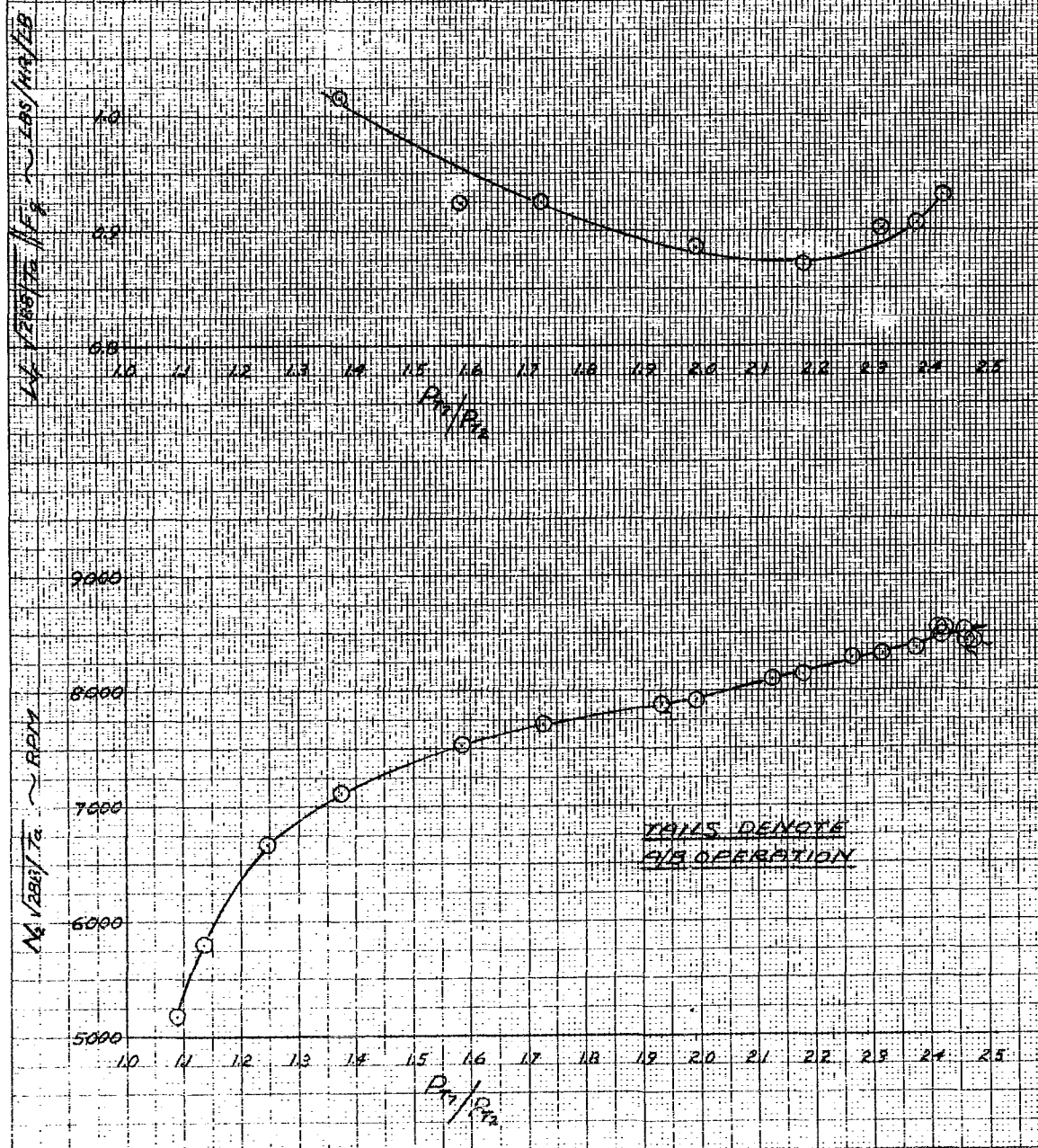


FIG. NO. 28
GROSS THRUST COEFFICIENT
ENGINE YJ-75-P-3 SIN 610016
YJ-75-P-3 SIN 610029

AFTERBURNING

FRONT PRATT & WHITNEY CURVE
NO. INST 18181 DATED 8-14-56

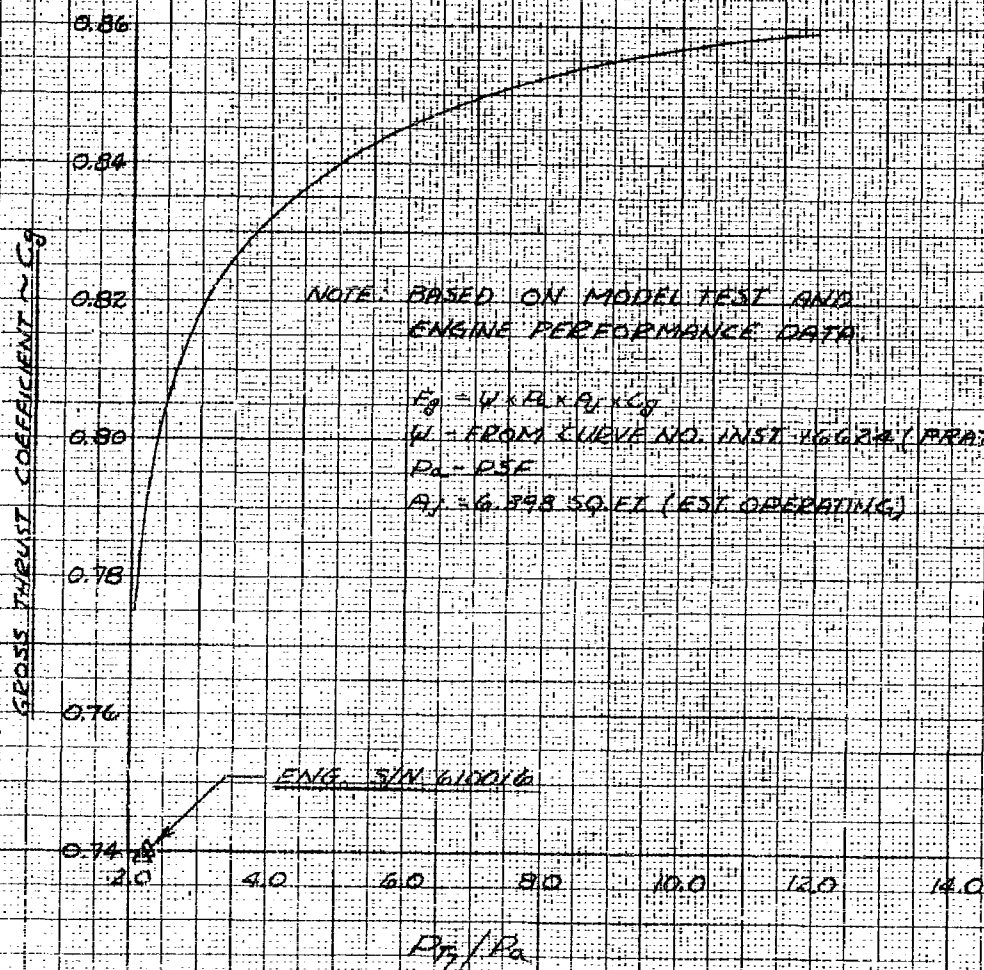


FIG. NO. 29
GROSS THRUST COEFFICIENT
ENGINE W-15-P-3 SIN 610016
W-15-P-3 SIN 610023

NON AFTERBURNING
FROM PRATT & WHITNEY CURVE
NO. INST. 27214 DATED 11-5-56

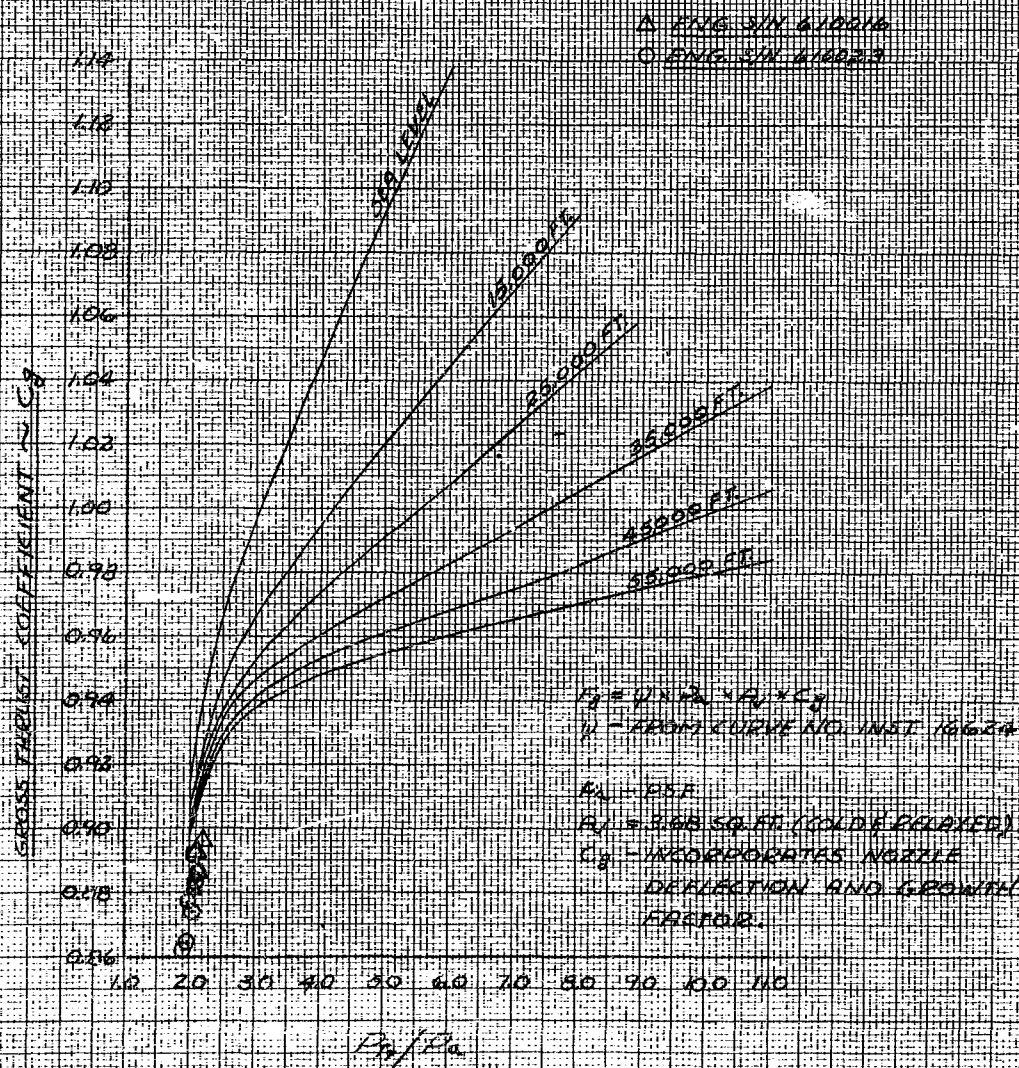


FIG No. 30
INLET DUCT RECOVERY
F-105R USAF NO. 54-100
MAXIMUM POWER

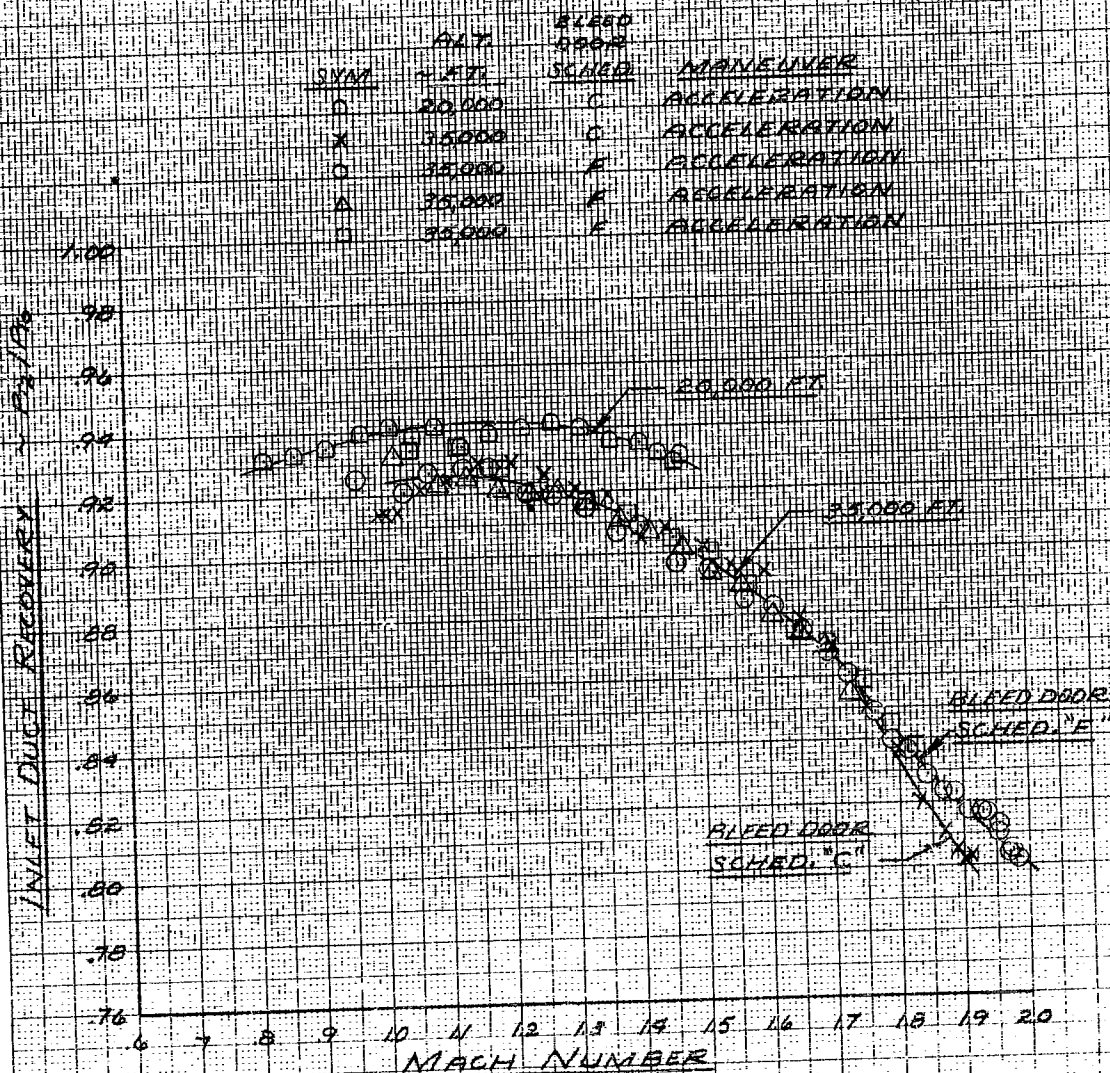


FIG. NO. 31
INLET DUCT PLUG SCHEDULE
F-105B USAF NO. 54-100
YJ-75-P-3 ENGINE

FROM R.A.D. DESIGN DATA

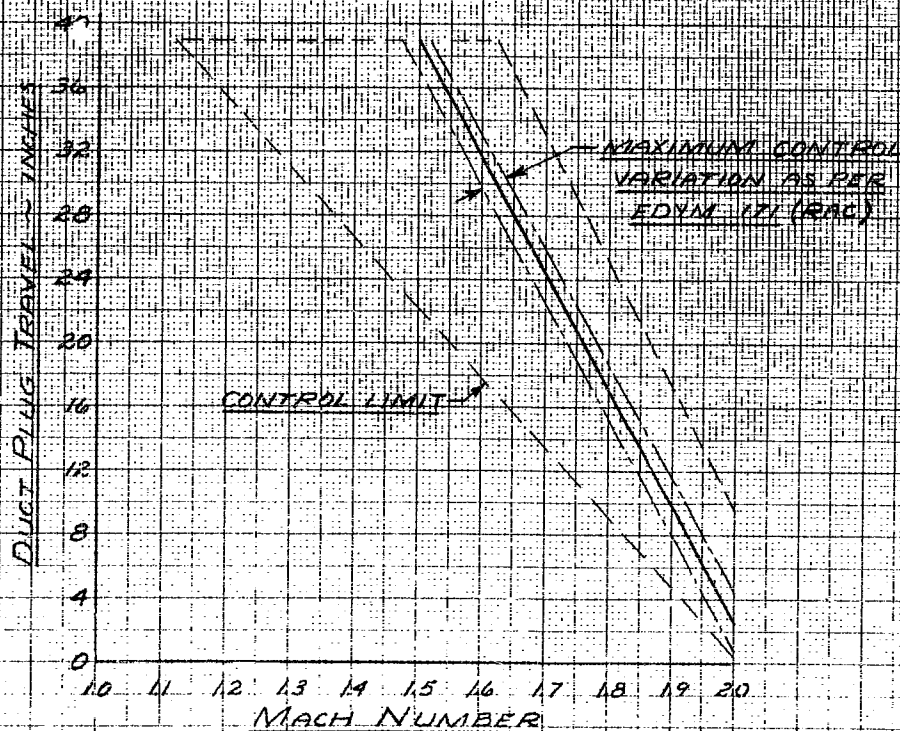


FIG No. 32
BLEED DOOR SCHEDULE 'C'
F-105B USAF NO. 59-100
IN-75 P-3 ENGINE

ALTITUDE 35,000 FT

FROM R.A.C. DESIGN DATA

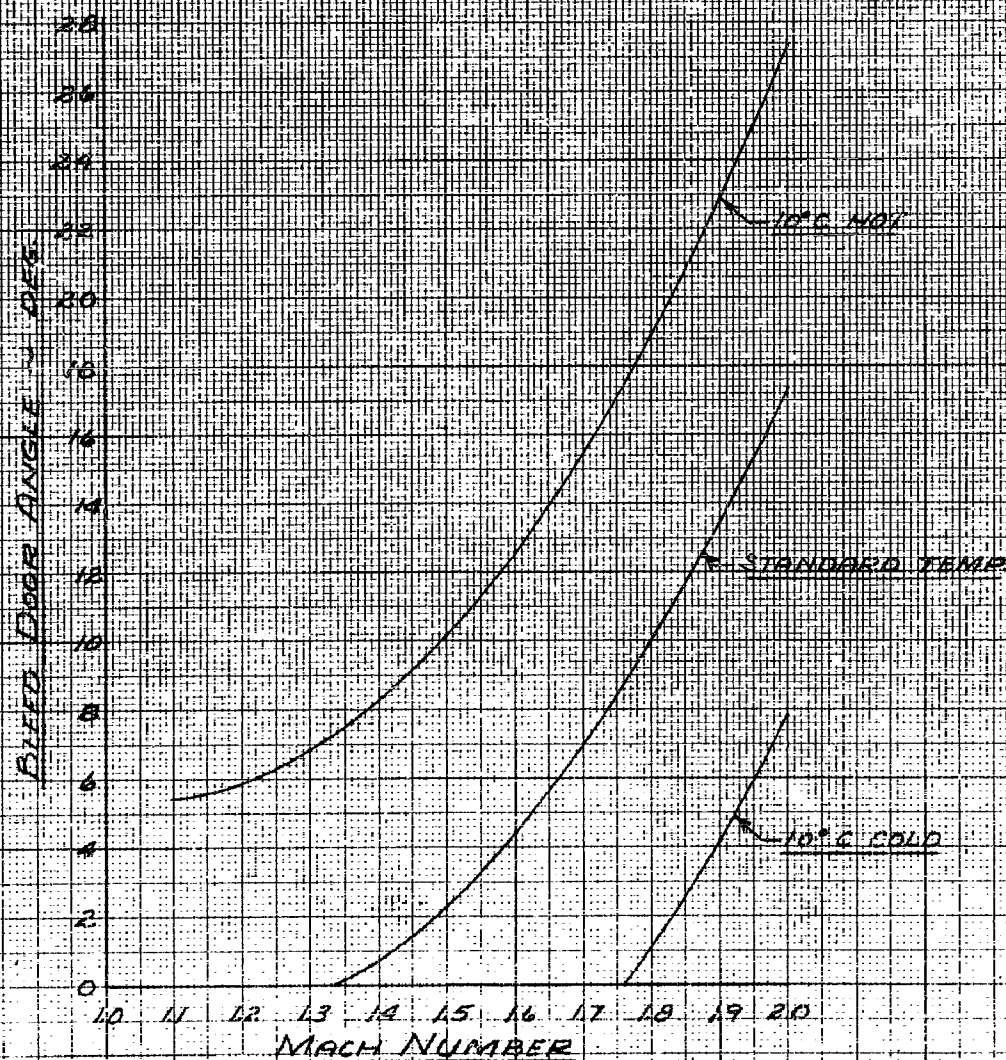


FIG. No. 83
BLEED DOOR SCHEDULE "F"
F-105B USAF NO. 54-100
Y1-75-P-3 ENGINE

ALTITUDE 35000 FT.

FROM R.A.C. DESIGN DATA

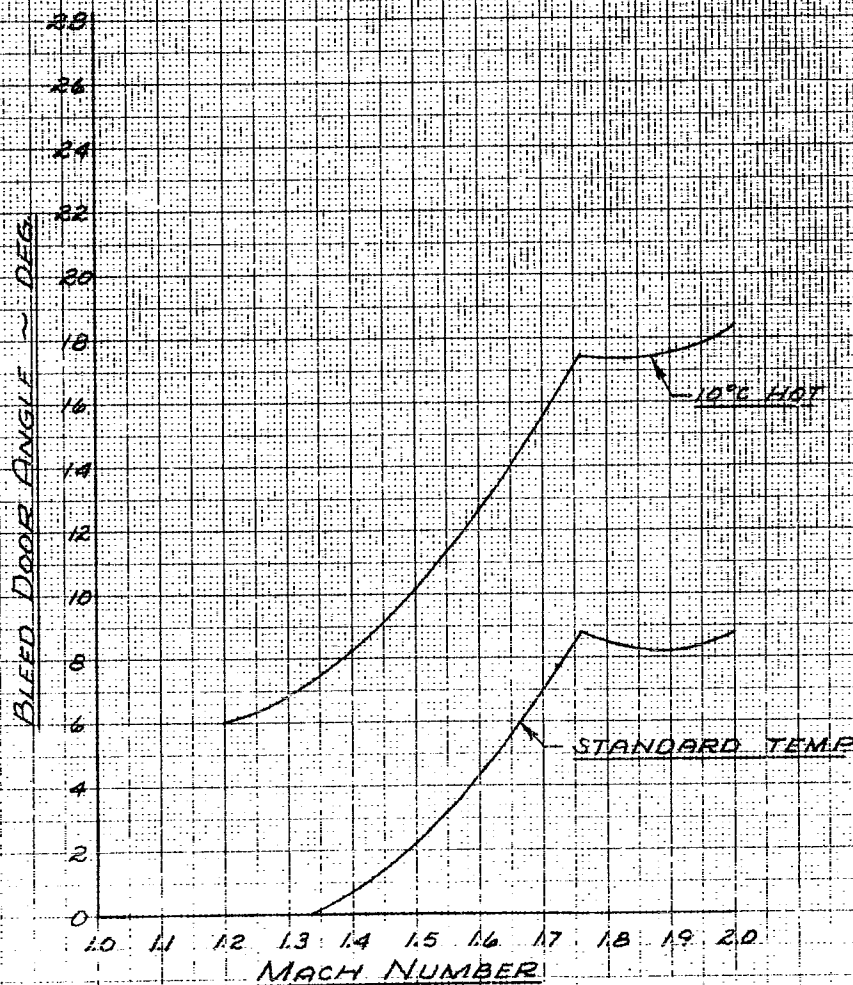


FIG. NO. 34
TRIM CURVE
F-105B USAF NO. 58-100
NO EXTERNAL STORES
CRUISE CONFIGURATION
C.G. POSITION 25.2 TO 26.3% MAC

LEADING EDGE FLAPS
SUBSONIC: 40% DOWN
SUPERSONIC: UP

O 20,000 FT. LEVEL ACCELERATION
Δ 20,000 FT. STABILIZED POINTS
□ 35,000 FT. LEVEL ACCELERATION
X 35,000 FT. LEVEL ACCELERATION

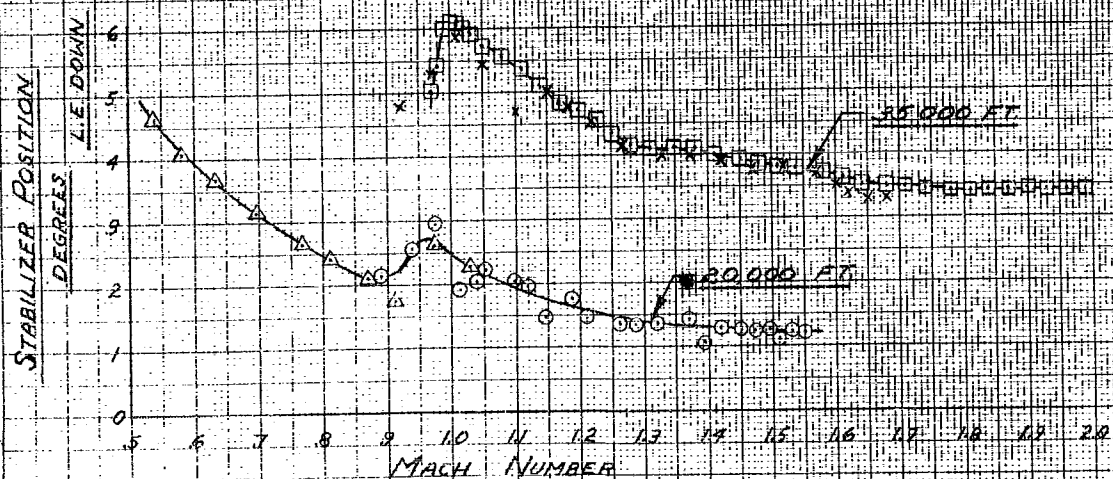


FIG. NO. 35
MANEUVERING FLIGHT
F-105B USAF NO. 54-100
GROSS WEIGHT 30,000 LBS.
C.G. POSITION 20.5% MAC
ALTITUDE 19,000 FT.
AIRSPEED 210 KNOTS/MS
POWER APPROACH CONFIGURATION

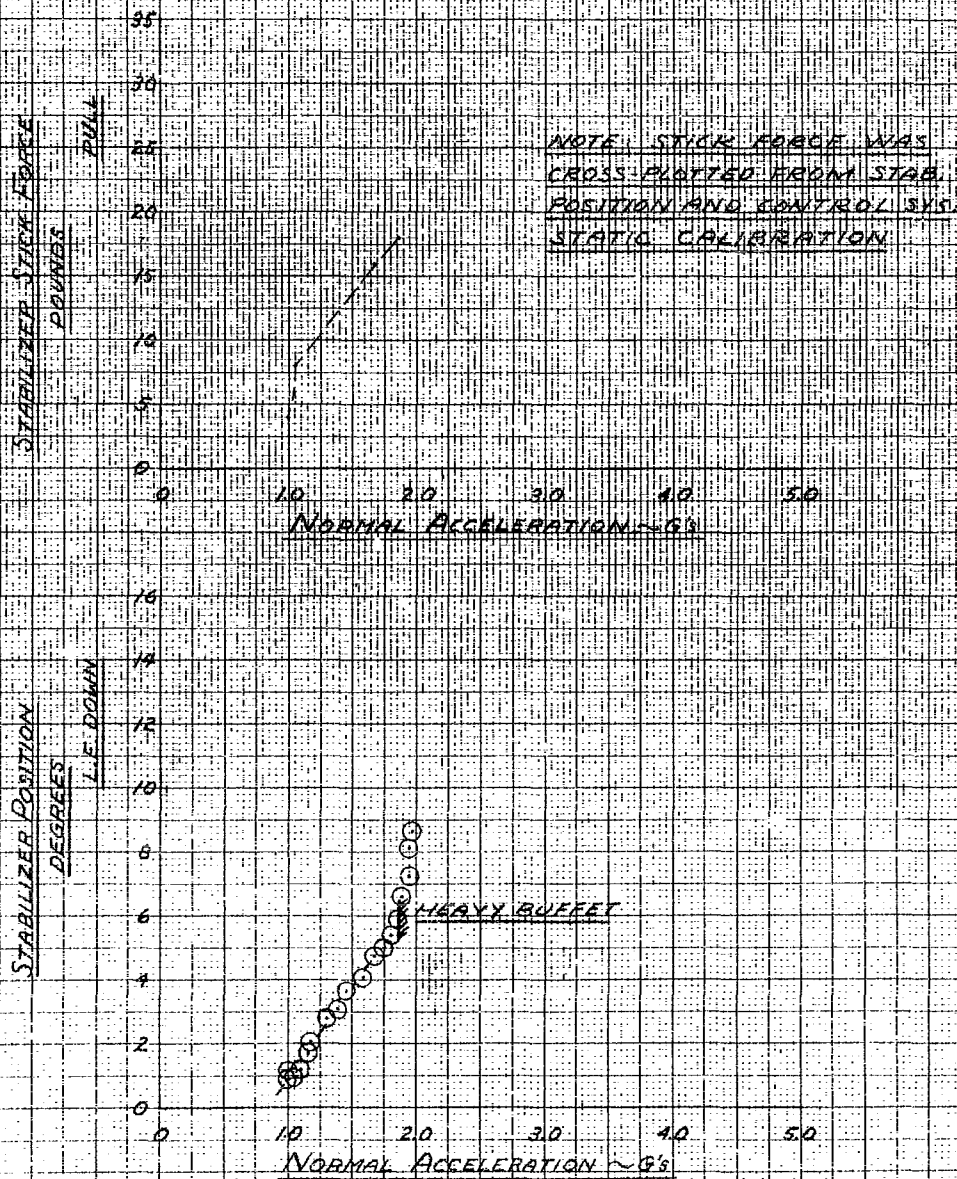


FIG. NO. 36
 MANEUVERING FLIGHT
 F-105B USAF NO. 54-100
 CRUISE CONFIGURATION
 GROSS WEIGHT 24,450 LBS.
 C.G. POSITION 26.2 MAC
 ALTITUDE 13,000 FT.
 MACH NO. .97

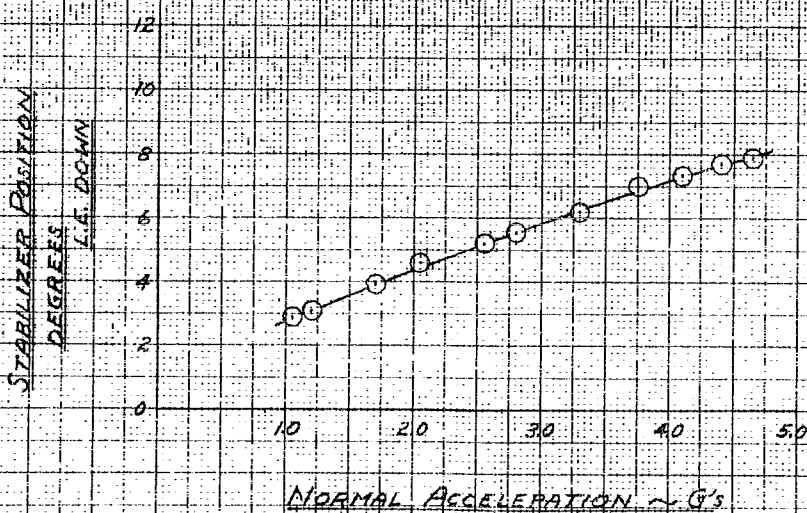
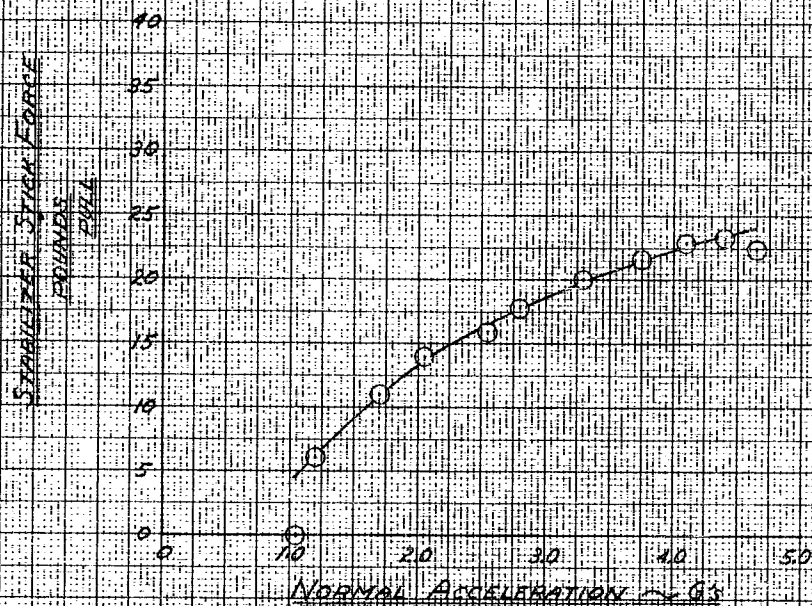


FIG. NO. 37
MANEUVERING FLIGHT
F-105B USAF NO. 54-100
CRUISE CONFIGURATION
GROSS WEIGHT 21,000 LBS.
C.G. POSITION 25.5% MAC
ALTITUDE 20,000 FT.
MACH NO. 1.20
LEADING EDGE FLAPS UP

NOTE: STICK FORCE WAS
OBTAINED FROM STABILIZER
POSITION AND CONTROL SYS.
STATIC CALIBRATION

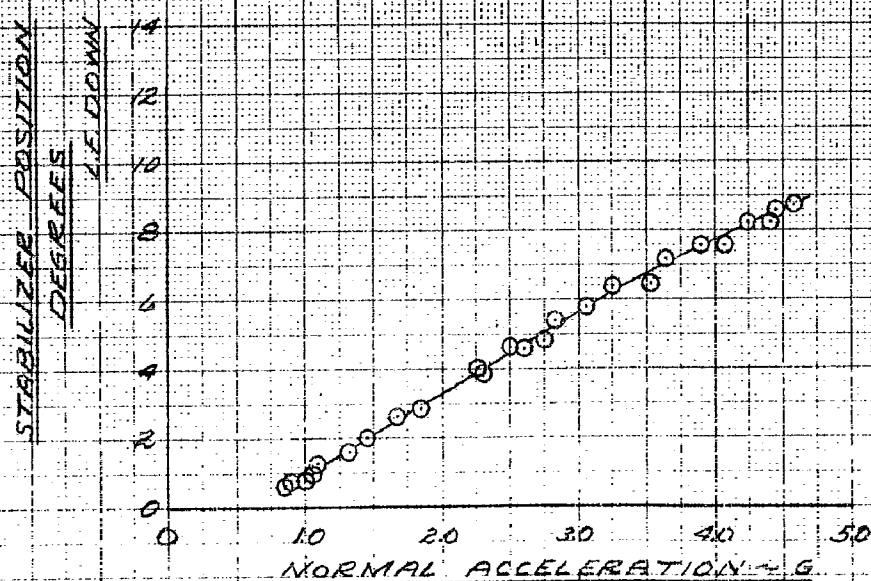
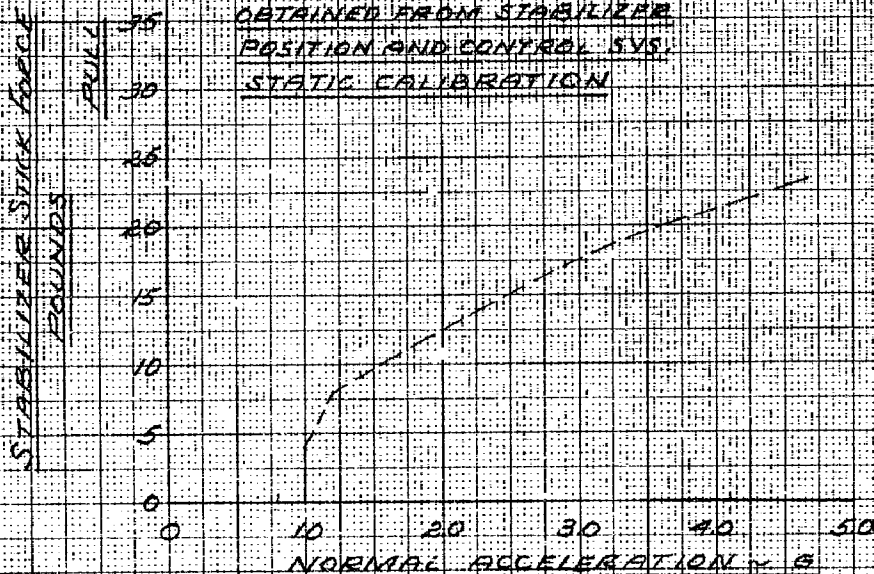


FIG. NO. 38
MANEUVERING FLIGHT
F-105B USAF NO. 54-100
CRUISE CONFIGURATION
GROSS WEIGHT 31,000 LBS.
C.G. POSITION 25.52 MAC
ALTITUDE 20,000 FT.
MACH NO. 1.42
LEADING EDGE FLAPS UP

NOTE: STICK FORCE WAS
OBTAINED FROM STABILIZER
POSITION AND CONTROL SYSTEM
STATIC CALIBRATION

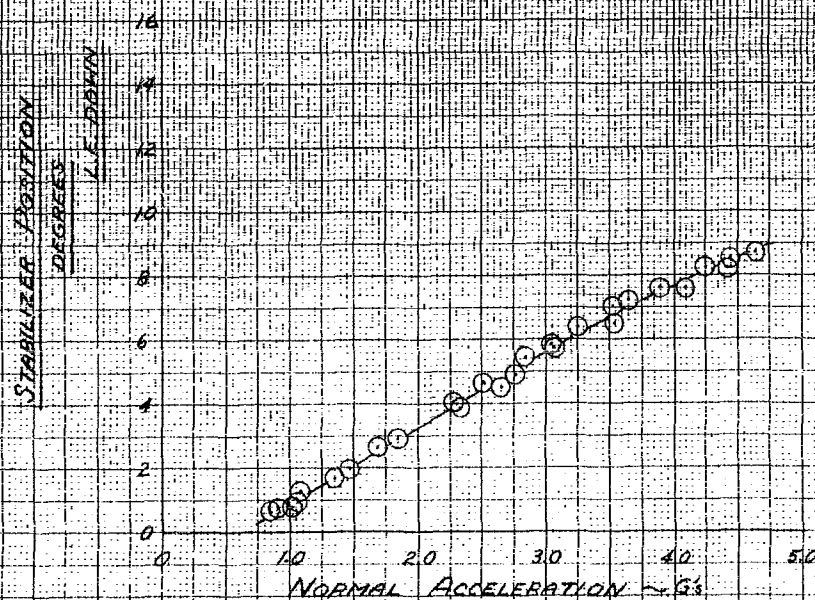
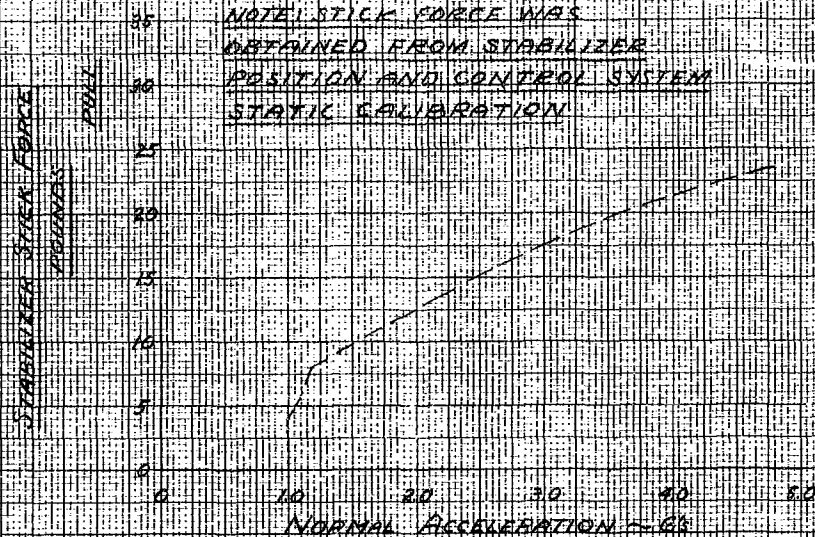


FIG. NO. 39
MANEUVERING FLIGHT
F-105B USAF NO. 54-100
CRUISE CONFIGURATION
GROSS WEIGHT 33,000 LBS
CG POSITION 25.8% M.A.C.
ALTITUDE 35,000 FT.
MACH NO. 0.93
LEADING EDGE FLAPS 38% DOWN

NOTE: STICK FORCE WAS
OBTAINED FROM STABILIZER
POSITION AND CONTROL SYSTEM
STATIC CALIBRATION

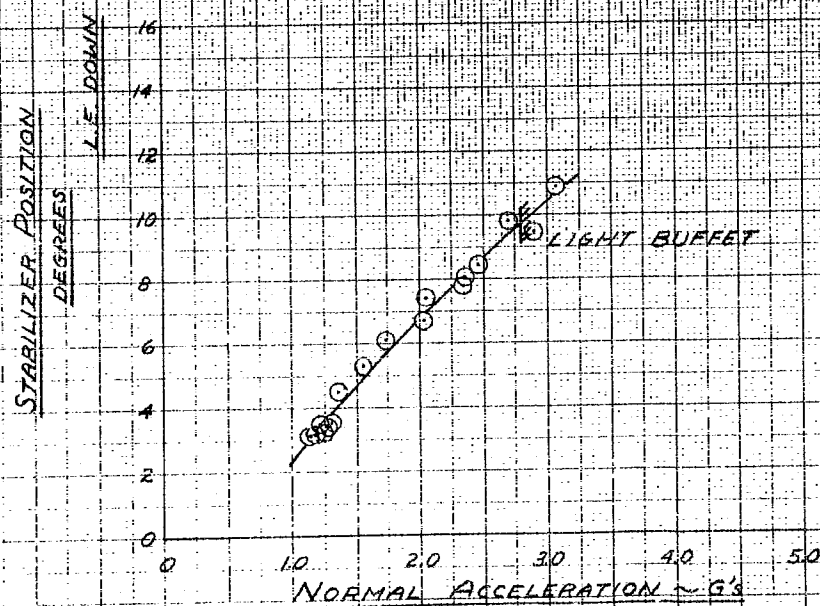
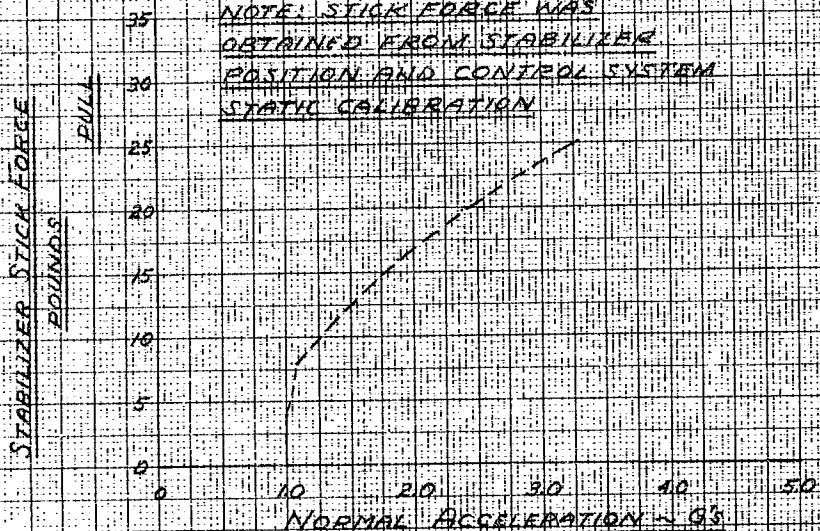


FIG. NO. 40
 MANEUVERING FLIGHT
 F-105B USAF NO. 54-100
 CRUISE CONFIGURATION
 GROSS WEIGHT 31,600 LBS.
 C.G. POSITION 25.4% MAC
 ALTITUDE 32,000 FT.
 MACH NO. 1.42
 LEADING EDGE FLAPS UP

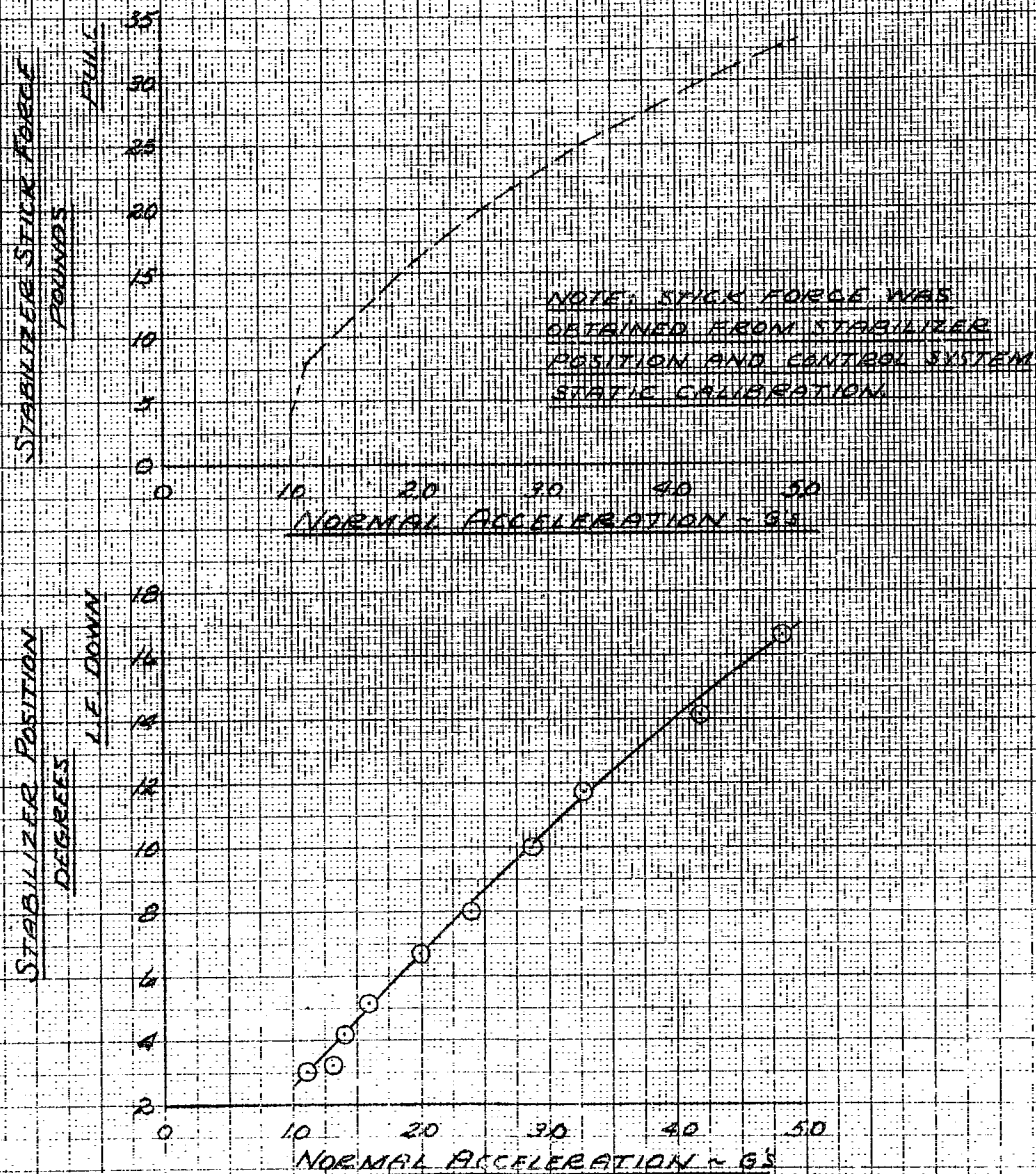


FIG No. 41
MANEUVERING FLIGHT
F-105B USAF NO. 54-100
CRUISE CONFIGURATION
GROSS WEIGHT 31,000 LBS
C.G. POSITION 25.5% MAC
ALTITUDE 35,000 FT.
MACH NO. 1.82
LEADING EDGE FLAPS UP

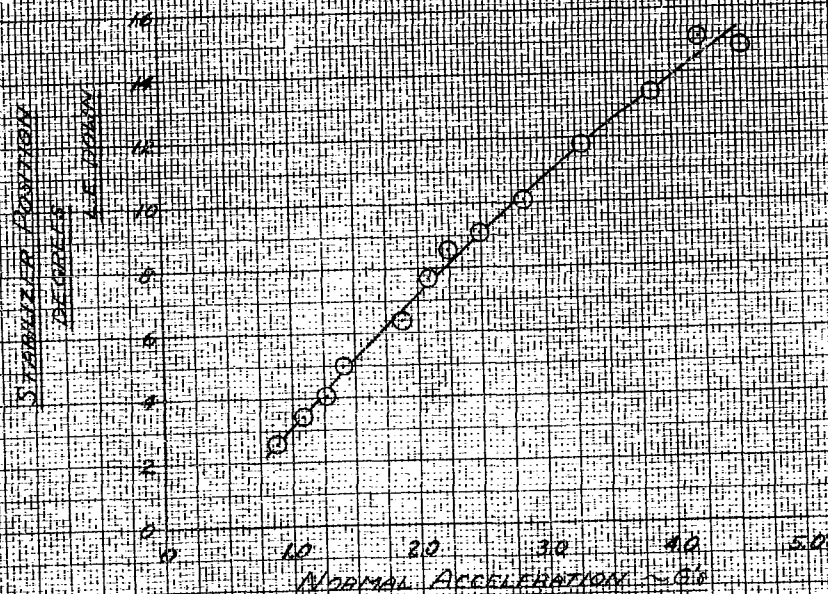
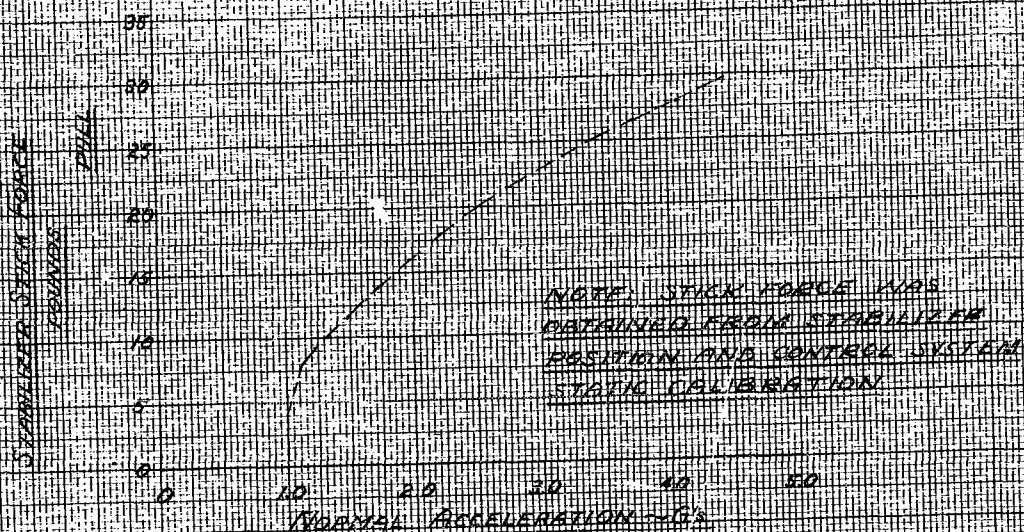


FIG. NO. 42
 MANEUVERING FLIGHT
 F-105B USAF NO. 88-100
 CRUISE CONFIGURATION
 GROSS WEIGHT 29,300 LBS
 CG POSITION 25.7% MAC
 ALTITUDE 35,000 FT
 MACH NO. 1.65
 LEADING EDGE FLAPS UP

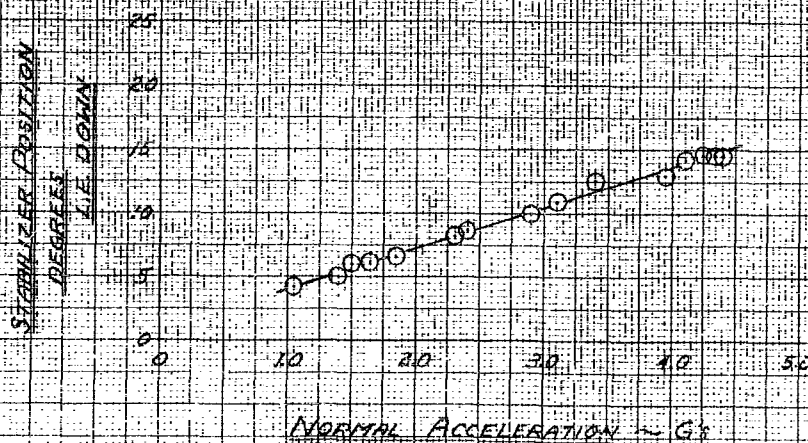
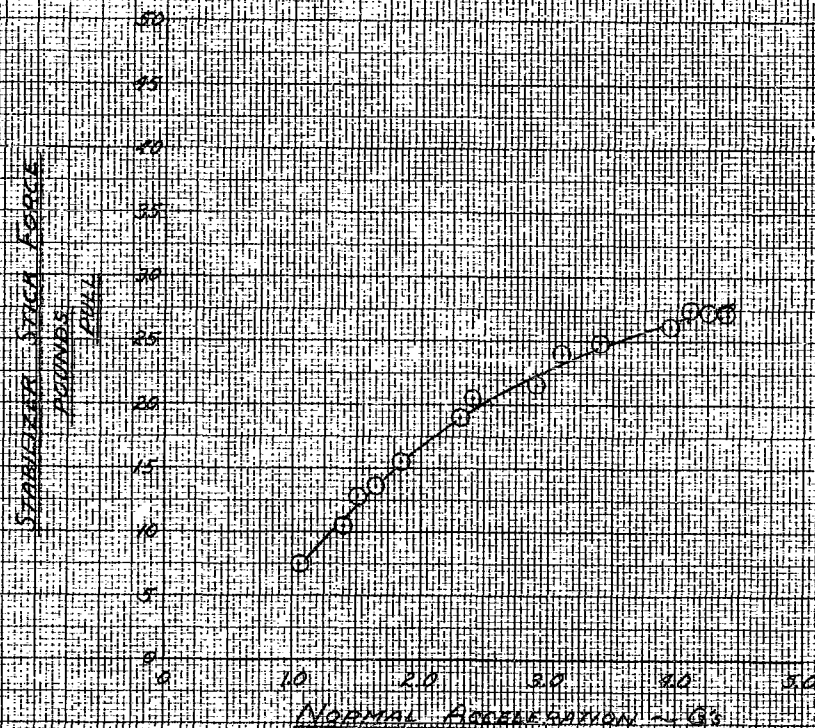


FIG. NO. 43
 MANEUVERING FLIGHT
 FLIGHTER USAF NO. 54-100
 CRUISE CONFIGURATION
 GROSS WEIGHT 29,450 LB
 C.G. POSITION 2.53 %MAC
 ALTITUDE 40,000 FT
 MACH NO. .98
 LEADING EDGE FLAPS 30% DOWN

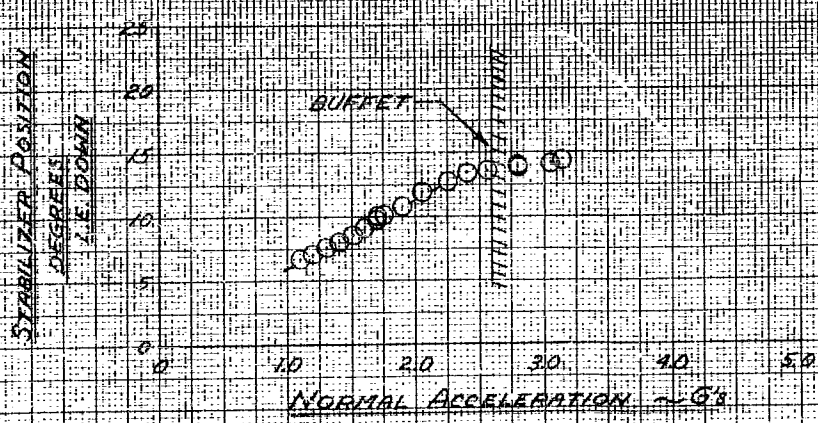
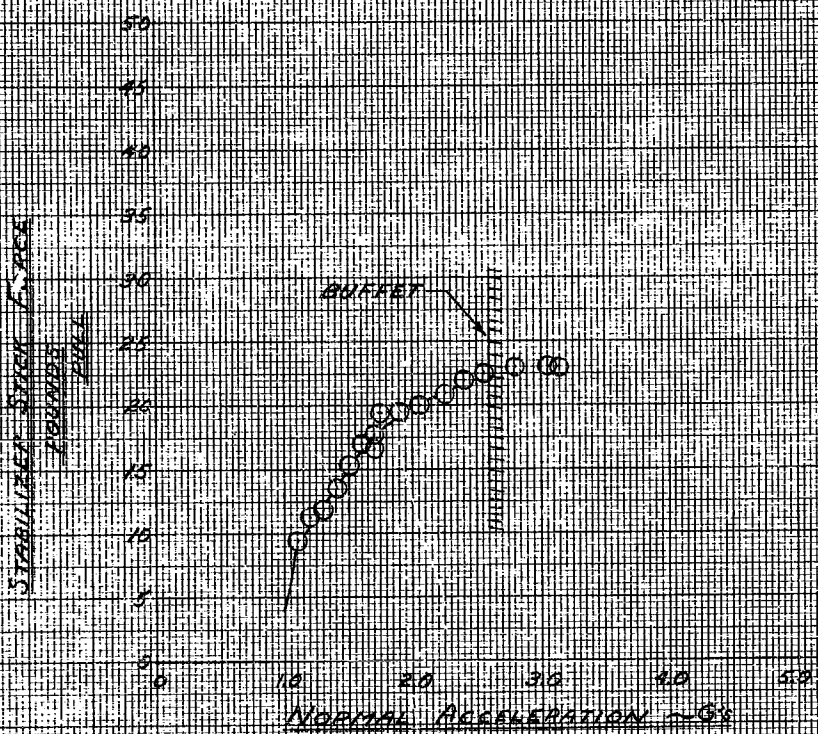


FIG. NO. 44
MANEUVERING FLIGHT
F-105B USAF NO. 59-100
ORLISE CONFIGURATION
GROSS WEIGHT 29,700 LBS.
C/G POSITION 25.2% MAC
ALTITUDE 43,500 LBS
MACH NO. 1.47
LEADING EDGE FLAPS UP

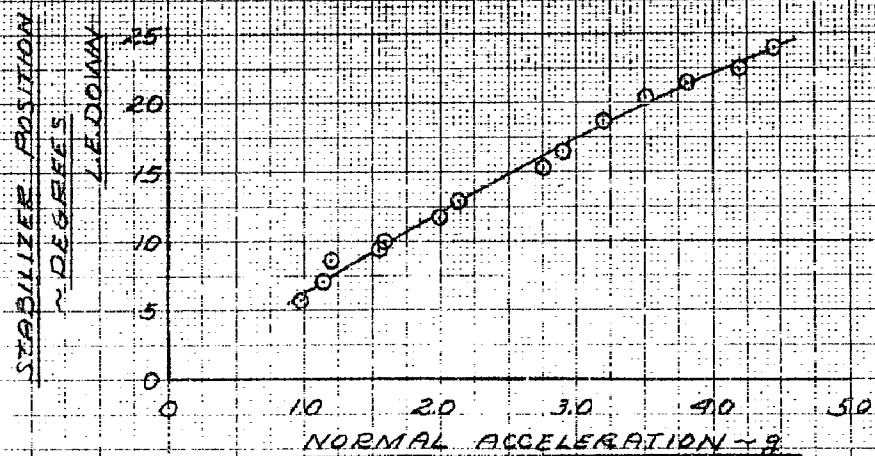
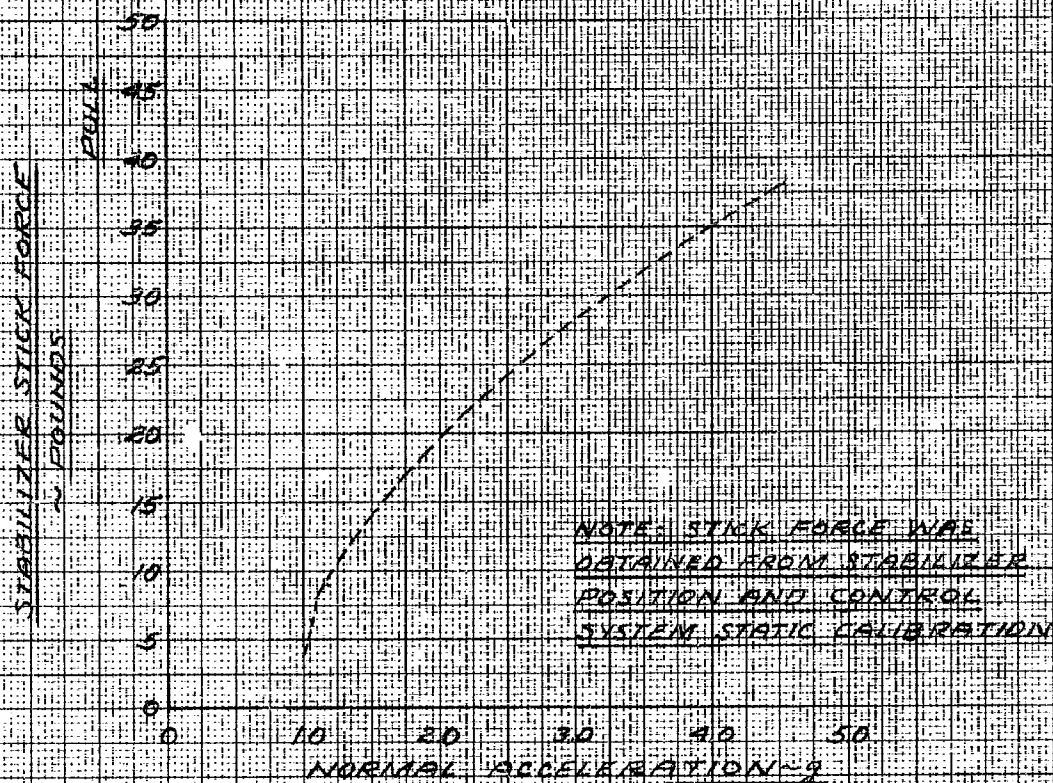


Figure No. 45
DYNAMIC LONGITUDINAL STABILITY
F-105B USAF No. 54-100

Cruise Configuration

Dampers On

Altitude 21,000 ft. Gross Weight 30,100 lbs.
Mach No. .908 CG 25.3 %MAC

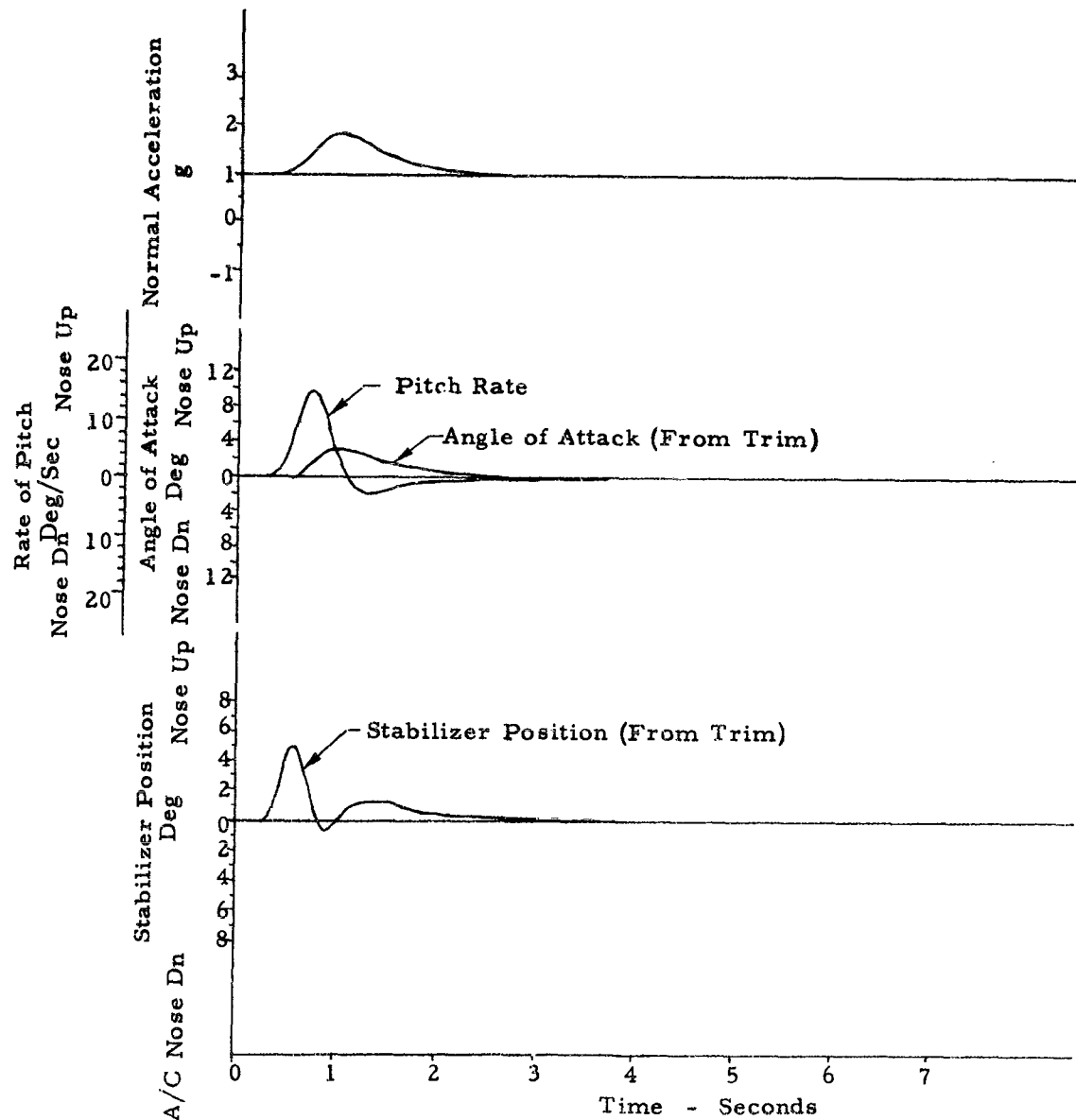


Figure No. 46
DYNAMIC LONGITUDINAL STABILITY
F-105B USAF No. 54-100
Cruise Configuration

Dampers Off
Altitude 35,800 ft. Gross Weight 32,950 lbs.
Mach No. .923 CG 25.9 %MAC

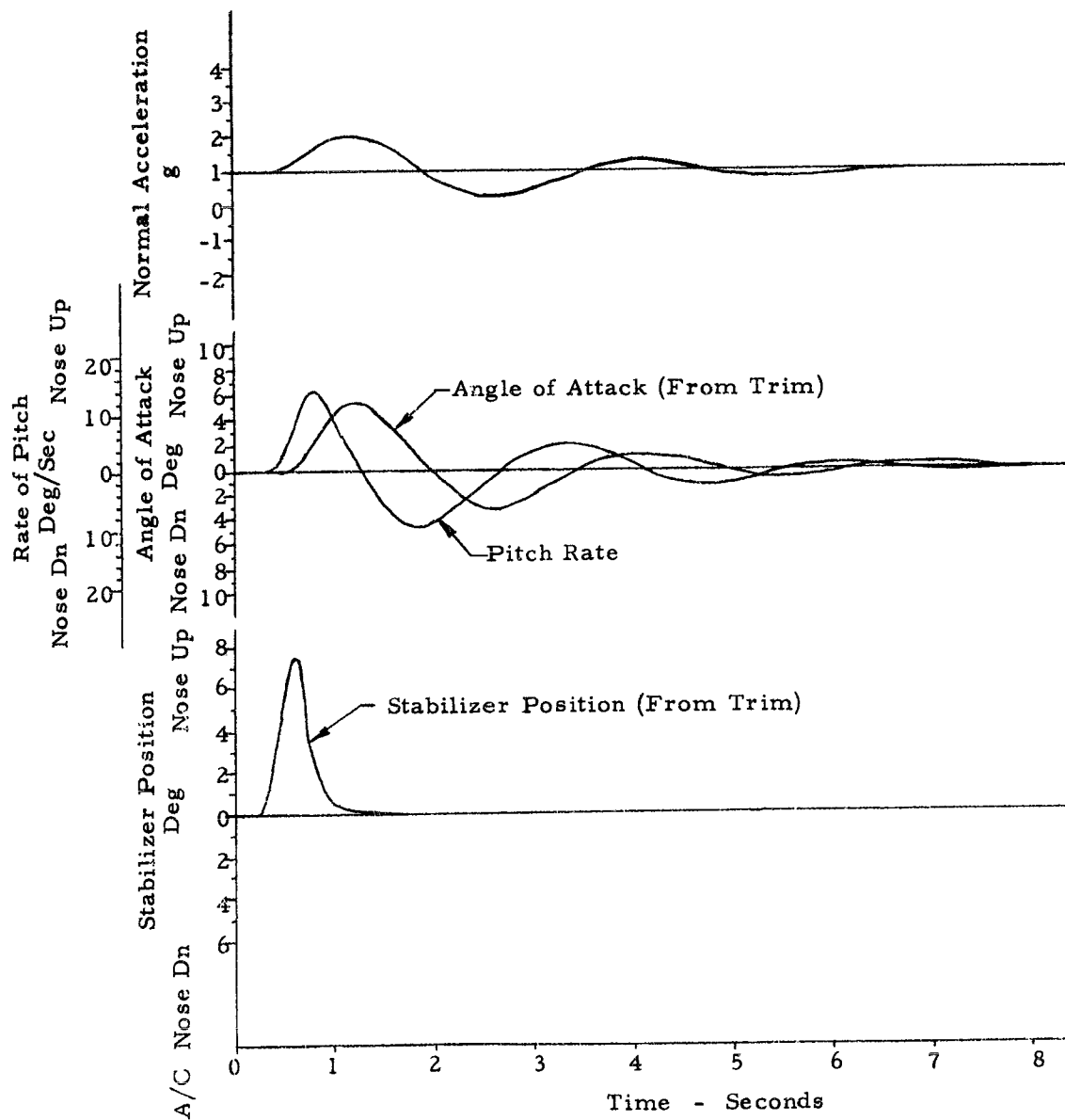


Figure No. 47
 DYNAMIC LONGITUDINAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration

Dampers Off
 Altitude 34,520 ft. Gross Weight 33,500 lbs.
 Mach No. 1.115 CG 26.0 %MAC

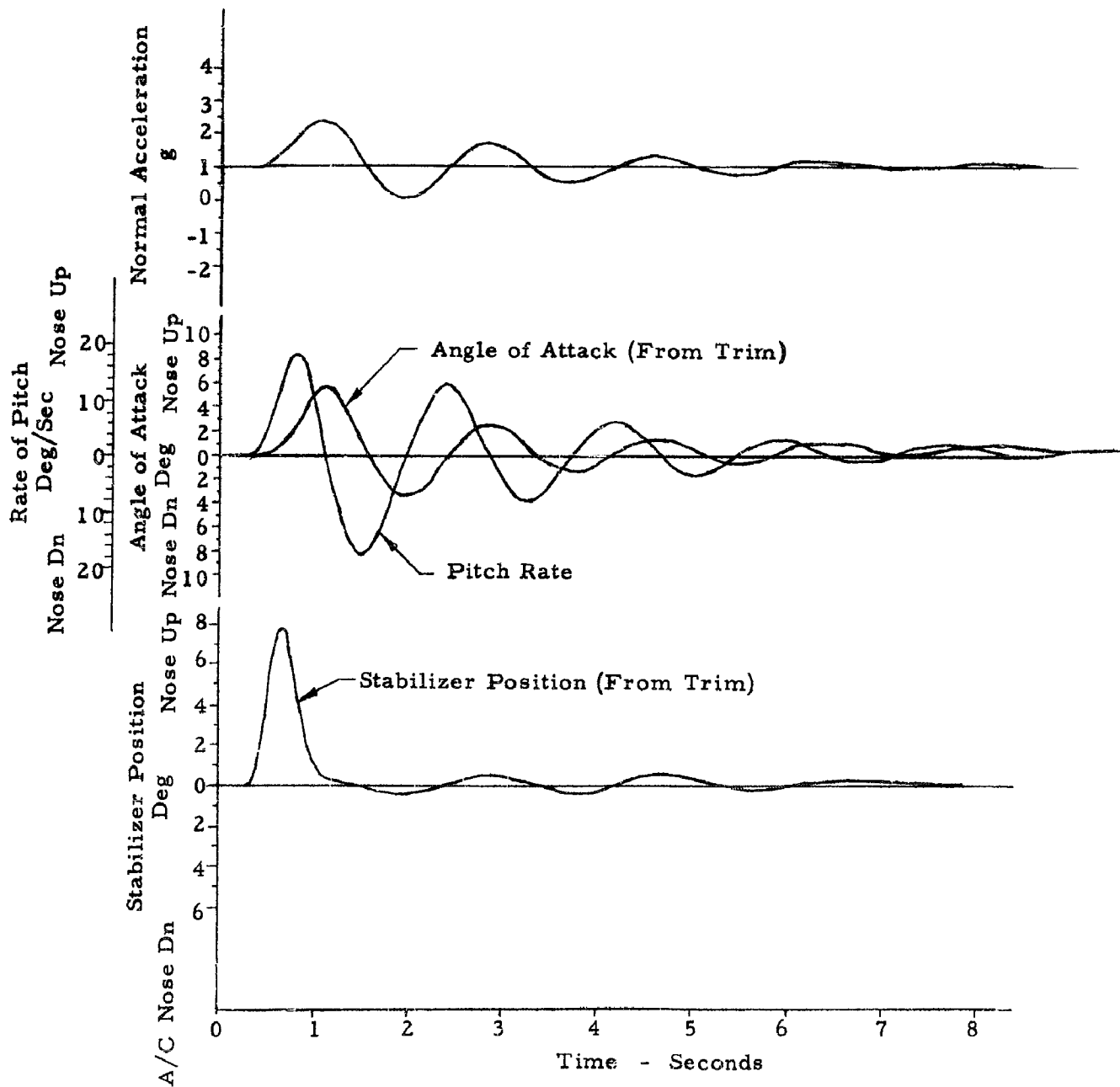


Figure No. 48
DYNAMIC LONGITUDINAL STABILITY
F-105B USAF No. 54-100

Cruise Configuration

Dampers On

Altitude 34,760 ft. Gross Weight 33,800 lbs.
Mach No. 1.122 CG 26.1 %MAC

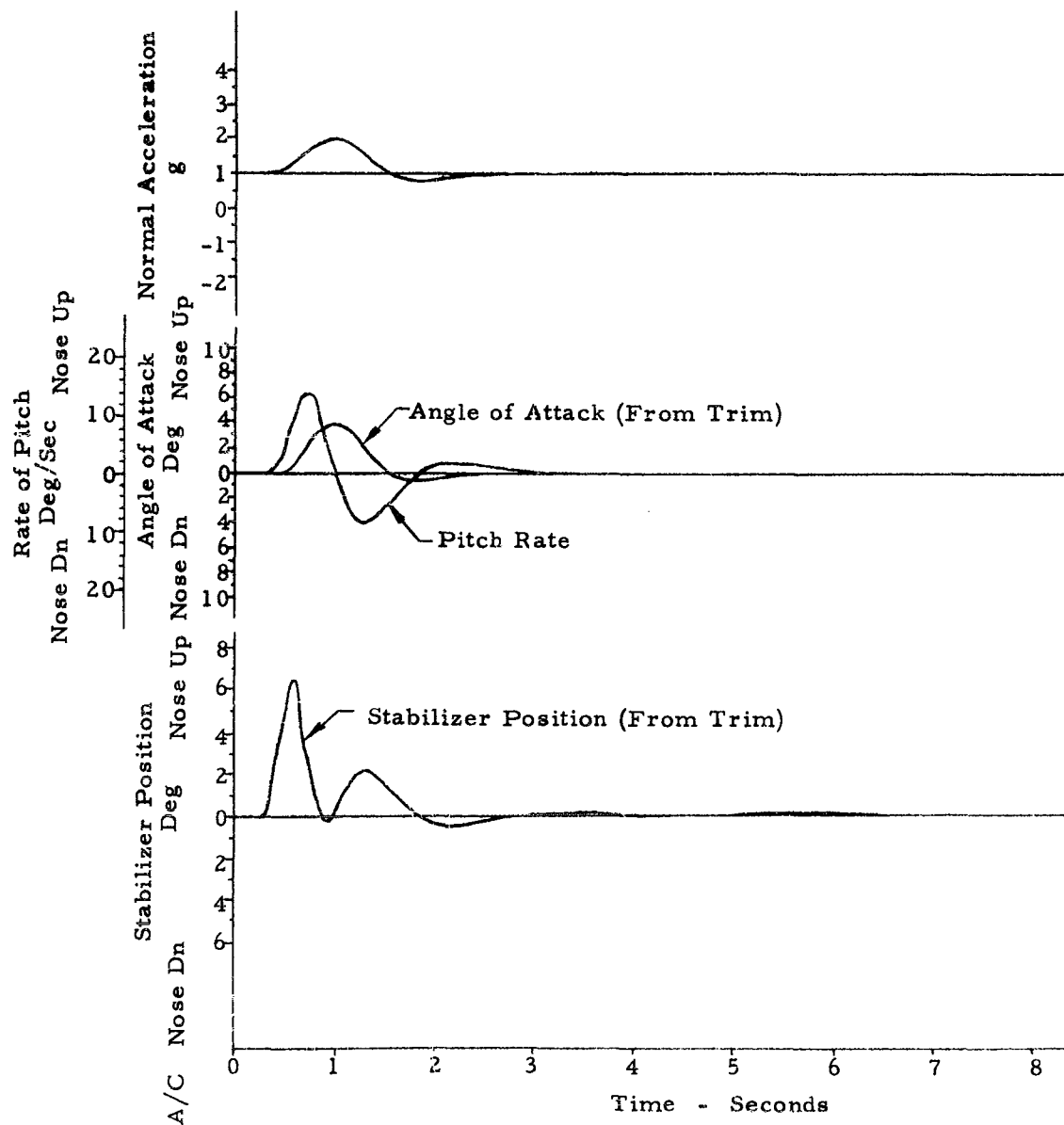


Figure No. 49
DYNAMIC LONGITUDINAL STABILITY
F-105B USAF No. 54-100
Cruise Configuration
Dampers On

Altitude 34,900 ft. Gross Weight 32,900 lbs.
Mach No. 1.72 CG 25.9% MAC

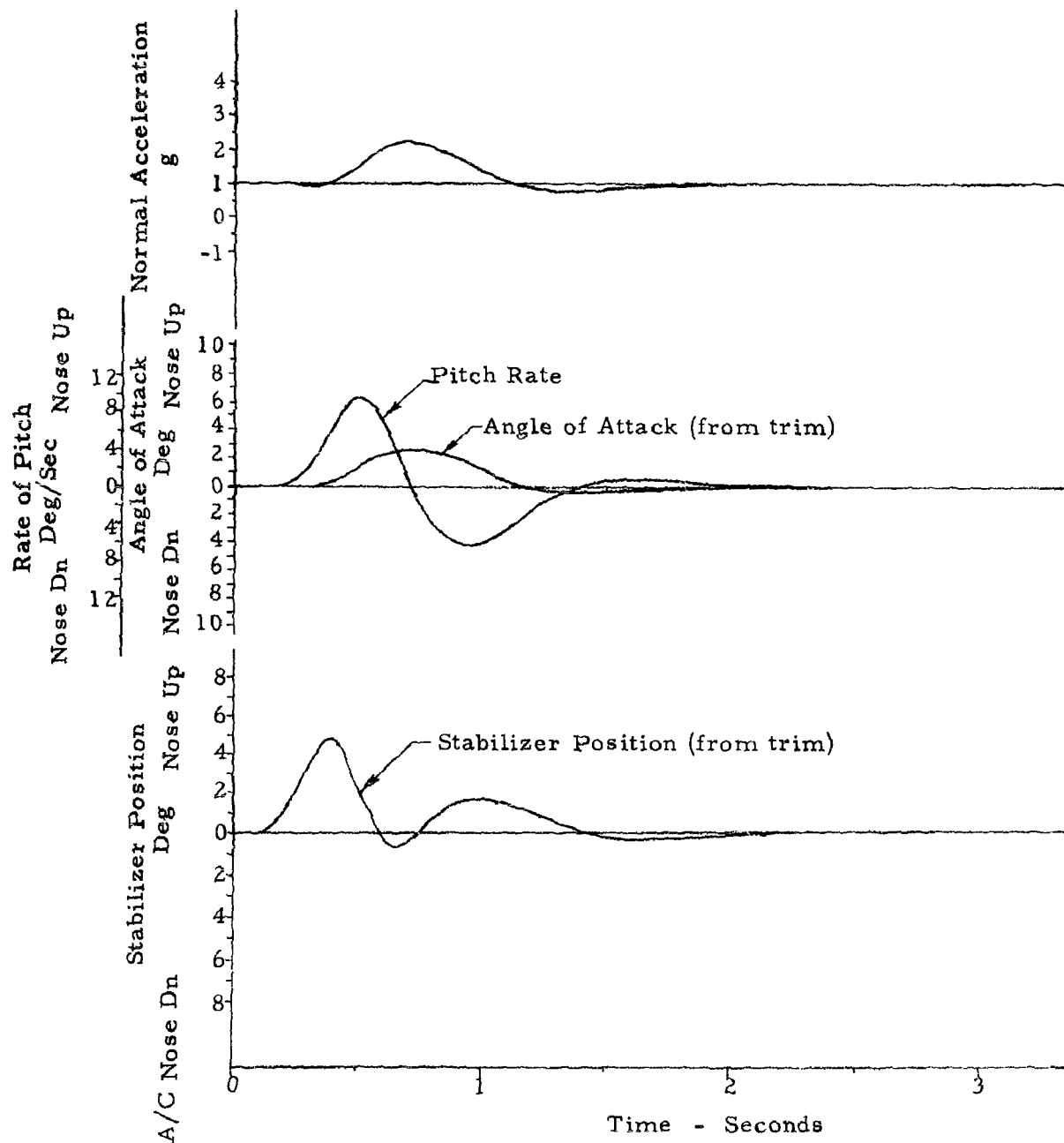


Figure No. 50
DYNAMIC LONGITUDINAL STABILITY
F-105B USAF No. 54-100

Cruise Configuration

Dampers On

Altitude 42,700 ft. Gross Weight 30,100 lbs.
Mach No. 1.40 CG 25.3% MAC

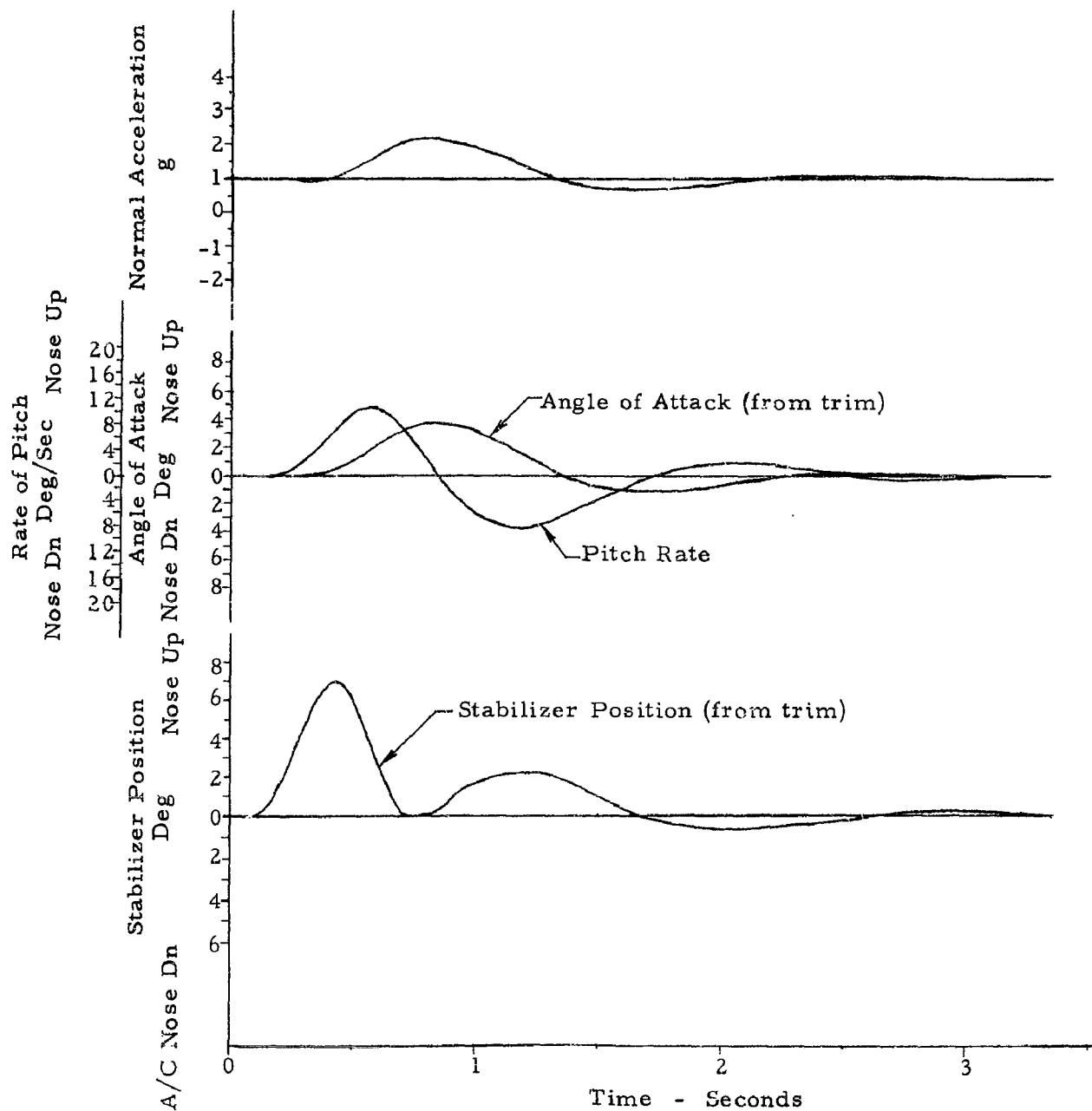


Figure No. 51
 DYNAMIC LONGITUDINAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 44,800 ft. Gross Weight 30,300 lbs.
 Mach No. 1.79 CG 25.3% MAC

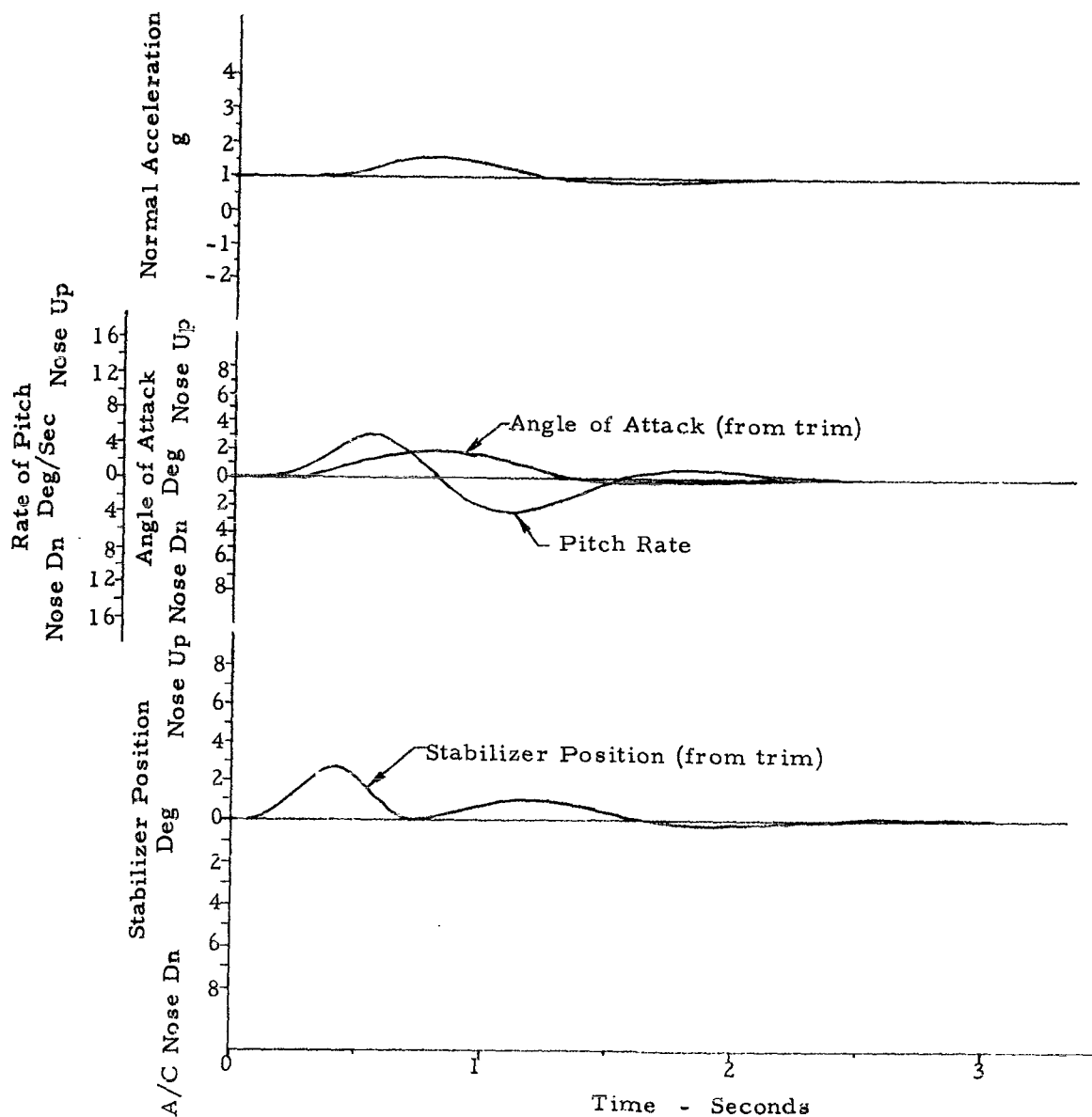


FIG NO 52
 STATIC DIRECTIONAL STABILITY
 F105A WEEF NO. 54100
 TRIM CONDITIONS
 CASE ~ 210 KNOTS
 ALT ~ 19000 FT
 WING NO. 160
 POWER APPROACH CONFIG.

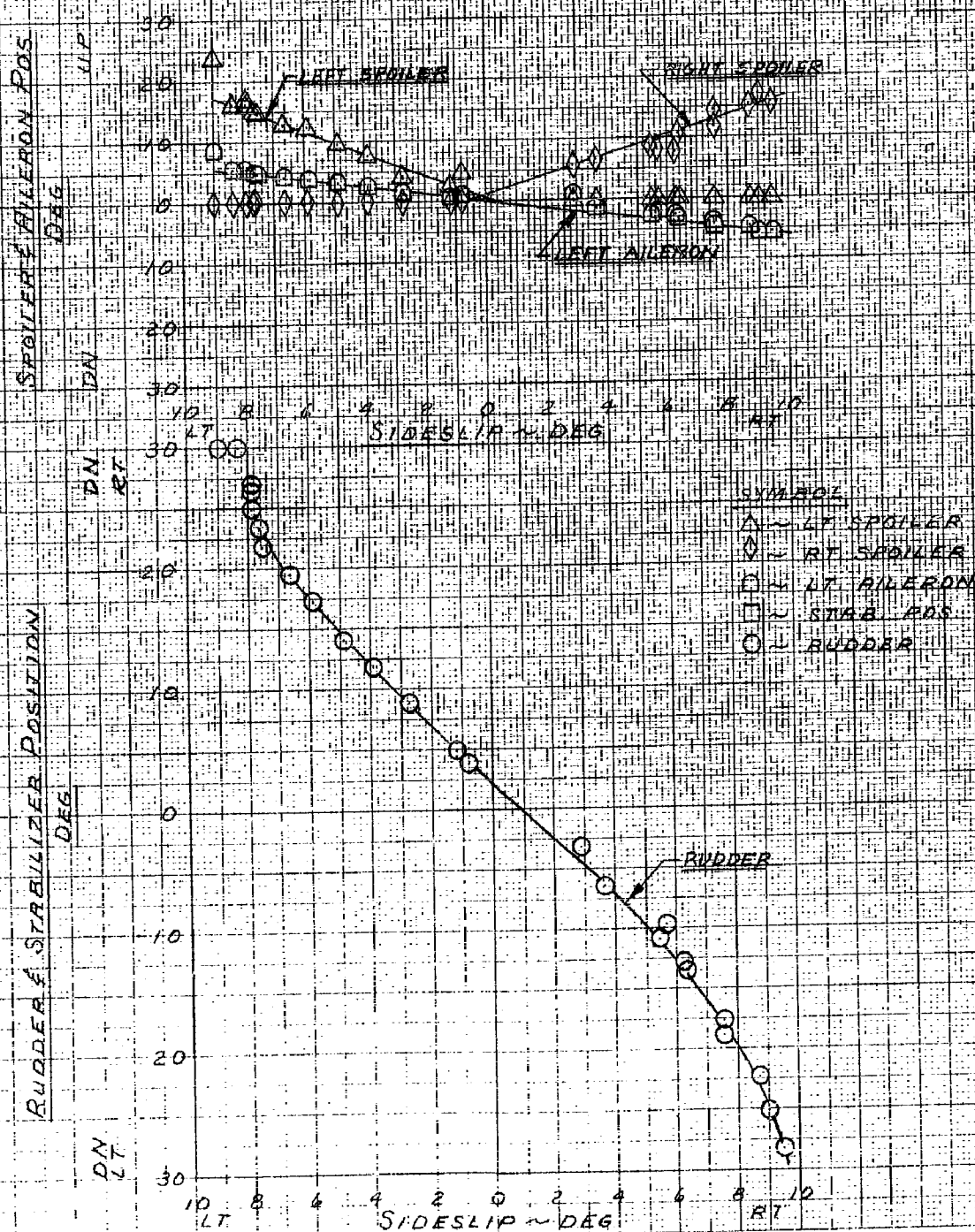


FIG NO. 53
STATIC DIRECTIONAL STABILITY
F-105B USAF NO. 59-100

TRIM CONDITIONS
CAS ~ 567 KNOTS
ALT ~ 81,700 FT
MACH NO. ~ 1.213

SPOLIER & AILERON POSITION

UP

DEG

DN

RUDDER & STABILIZER POS

UP

DEG

DN

LT

10

8

6

4

2

0

-2

-4

-6

-8

-10

-12

-14

-16

-18

-20

-22

-24

-26

-28

-30

-32

-34

-36

-38

-40

-42

-44

-46

-48

-50

-52

-54

-56

-58

-60

-62

-64

-66

10

8

6

4

2

0

-2

-4

-6

-8

-10

-12

-14

-16

-18

-20

-22

-24

-26

-28

-30

-32

-34

-36

-38

-40

-42

-44

-46

-48

-50

-52

-54

-56

-58

-60

-62

-64

-66

10

8

6

4

2

0

-2

-4

-6

-8

-10

-12

-14

-16

-18

-20

-22

-24

-26

-28

-30

-32

-34

-36

-38

-40

-42

-44

-46

-48

-50

-52

-54

-56

-58

-60

-62

-64

-66

10

8

6

4

2

0

-2

-4

-6

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-54

-56

-58

-60

-62

-64

-66

10

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6

4

2

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-4

-6

-8

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-12

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-48

-50

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-54

-56

-58

-60

-62

-64

-66

10

8

6

4

2

0

-2

-4

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-48

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-52

-54

-56

-58

-60

-62

-64

-66

10

8

6

4

2

0

-2

-4

-6

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-52

-54

-56

-58

-60

-62

-64

-66

10

8

6

4

2

0

-2

-4

-6

-8

-10

-12

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-16

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-22

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-50

-52

FIG. NO. 34
 STATIC DIRECTIONAL STABILITY
 F-105A USAF NO. 59-100
 TRIM CONDITIONS
 CAS ~ 318.5 KNOTS
 ALT ~ 35030 FT
 MACH NO. ~ 1.928

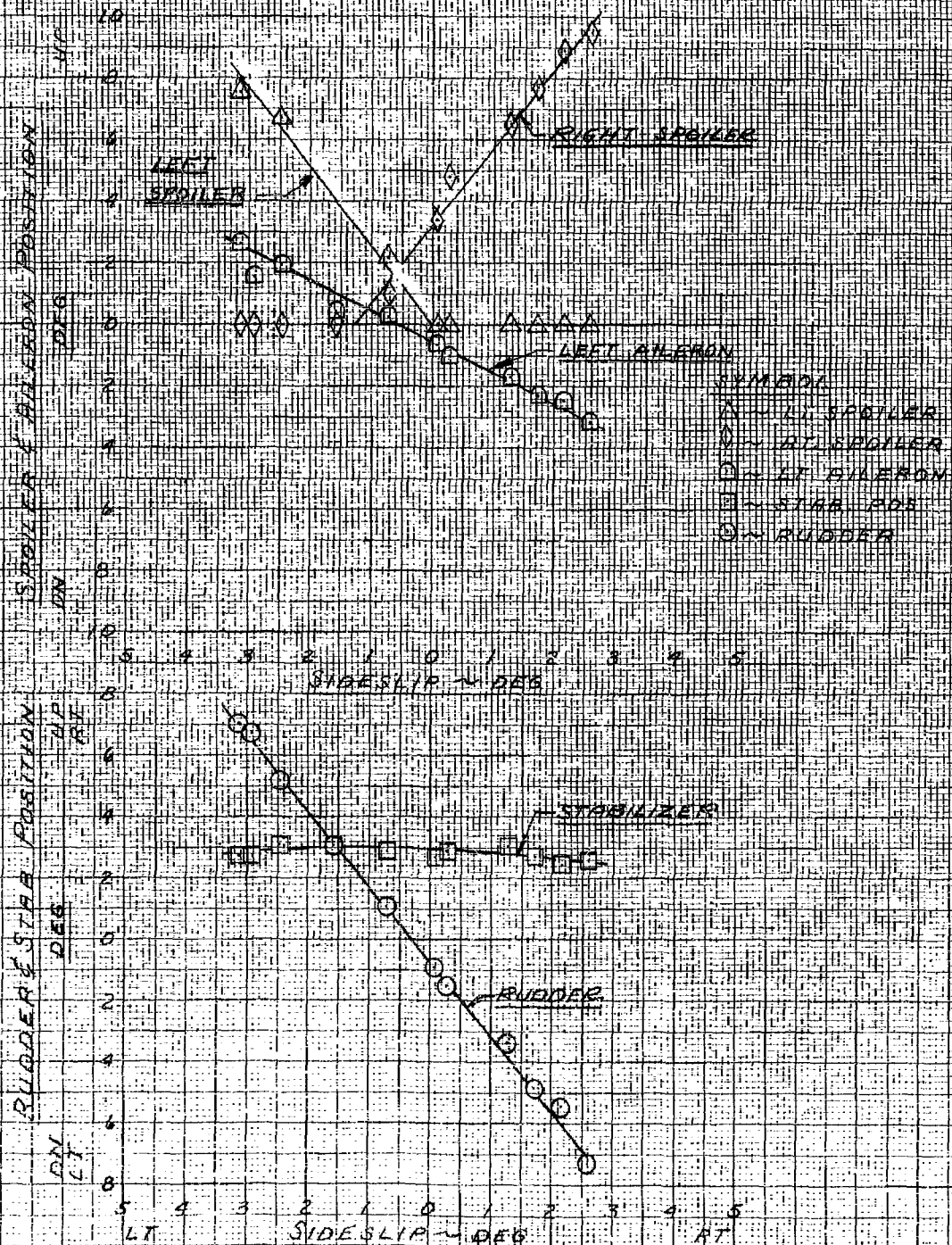
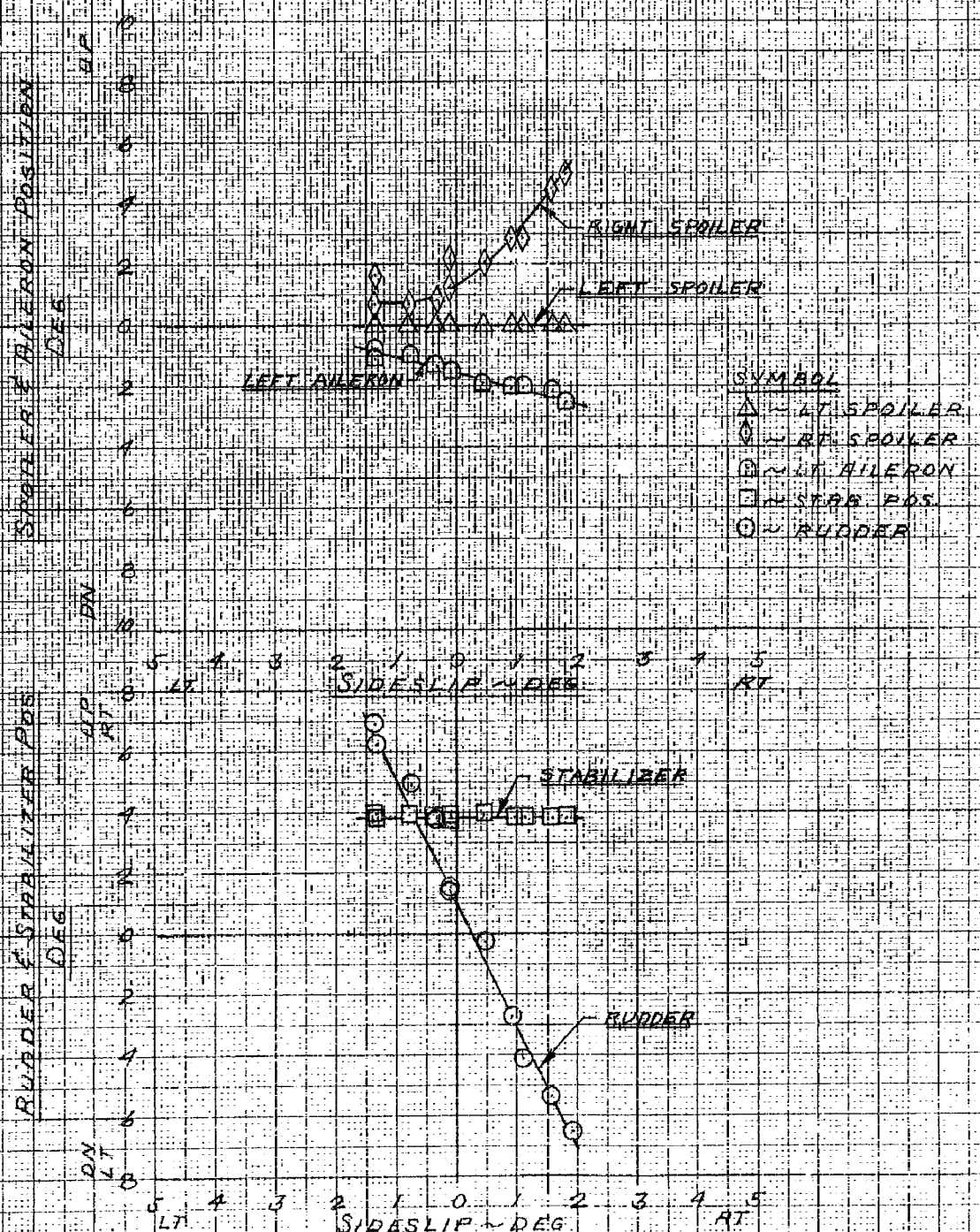


FIG. NO. 55
 STATIC DIRECTIONAL STABILITY
 F-105B USRF NO. 54-190
 TRIM CONDITIONS
 CASE 4015 KNOTS
 ALT 34,700 FT
 MACH NO. 1.18



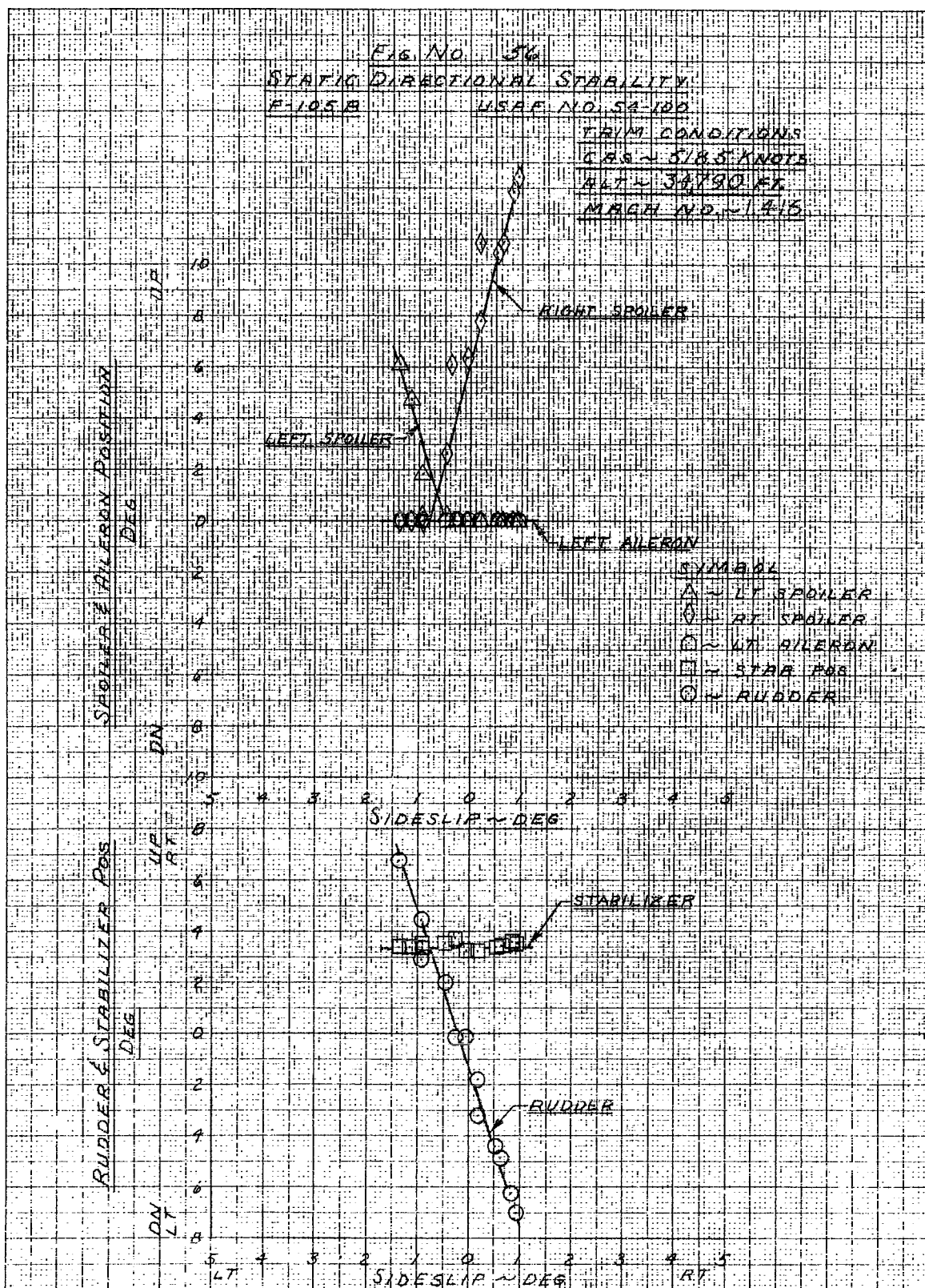


FIG NO. 51
 STATIC DIRECTIONAL STABILITY
 F-105B WAF NO. 54-100
 TRIM CONDITIONS
 CAS ~ 495 KTS
 ALT ~ 43 500 FT
 MACH NO ~ 1.43

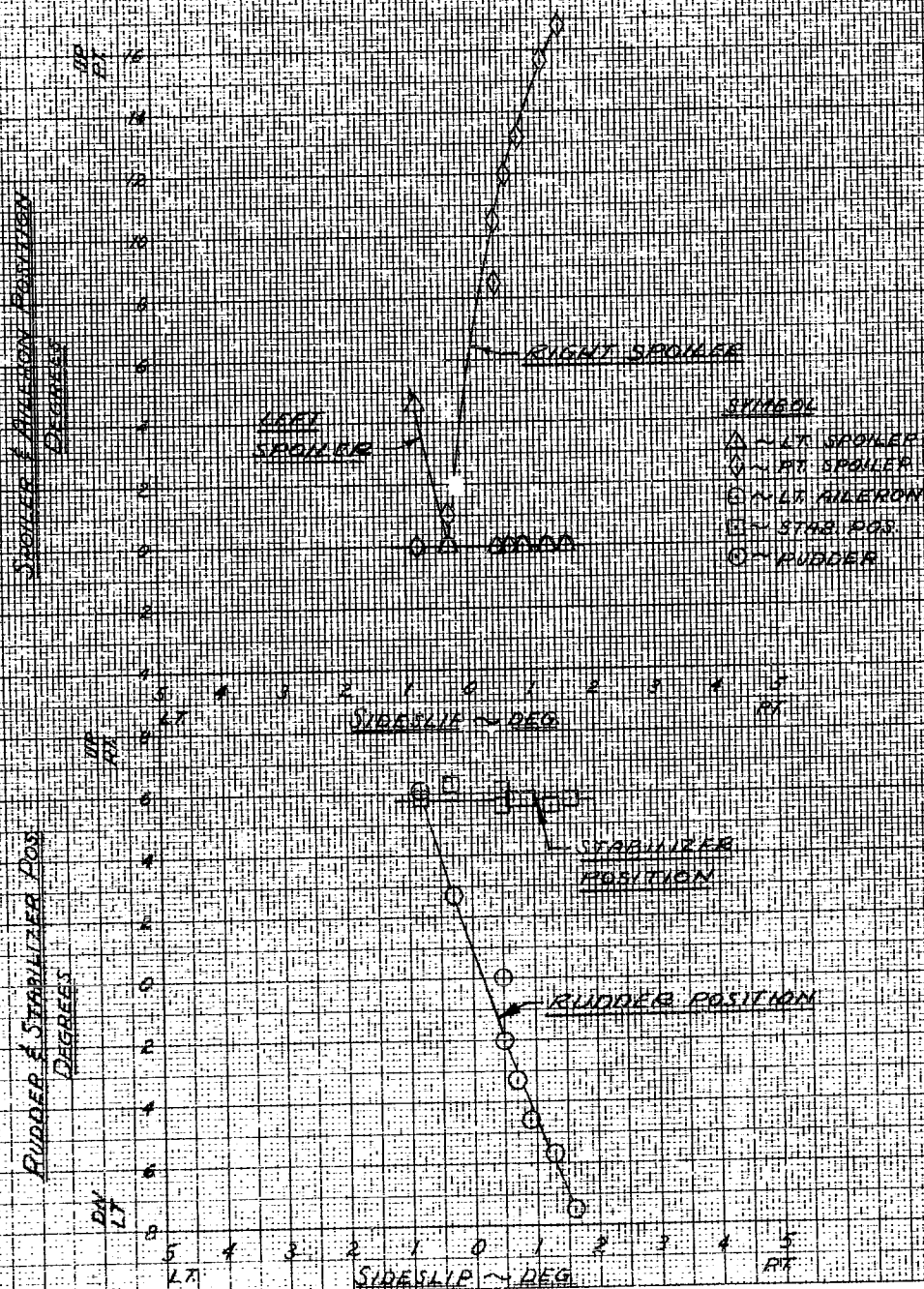


Figure No. 58
 DYNAMIC LATERAL-DIRECTIONAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 35,060 ft.
 Mach No. .932
 Pitch Damper Inoperative

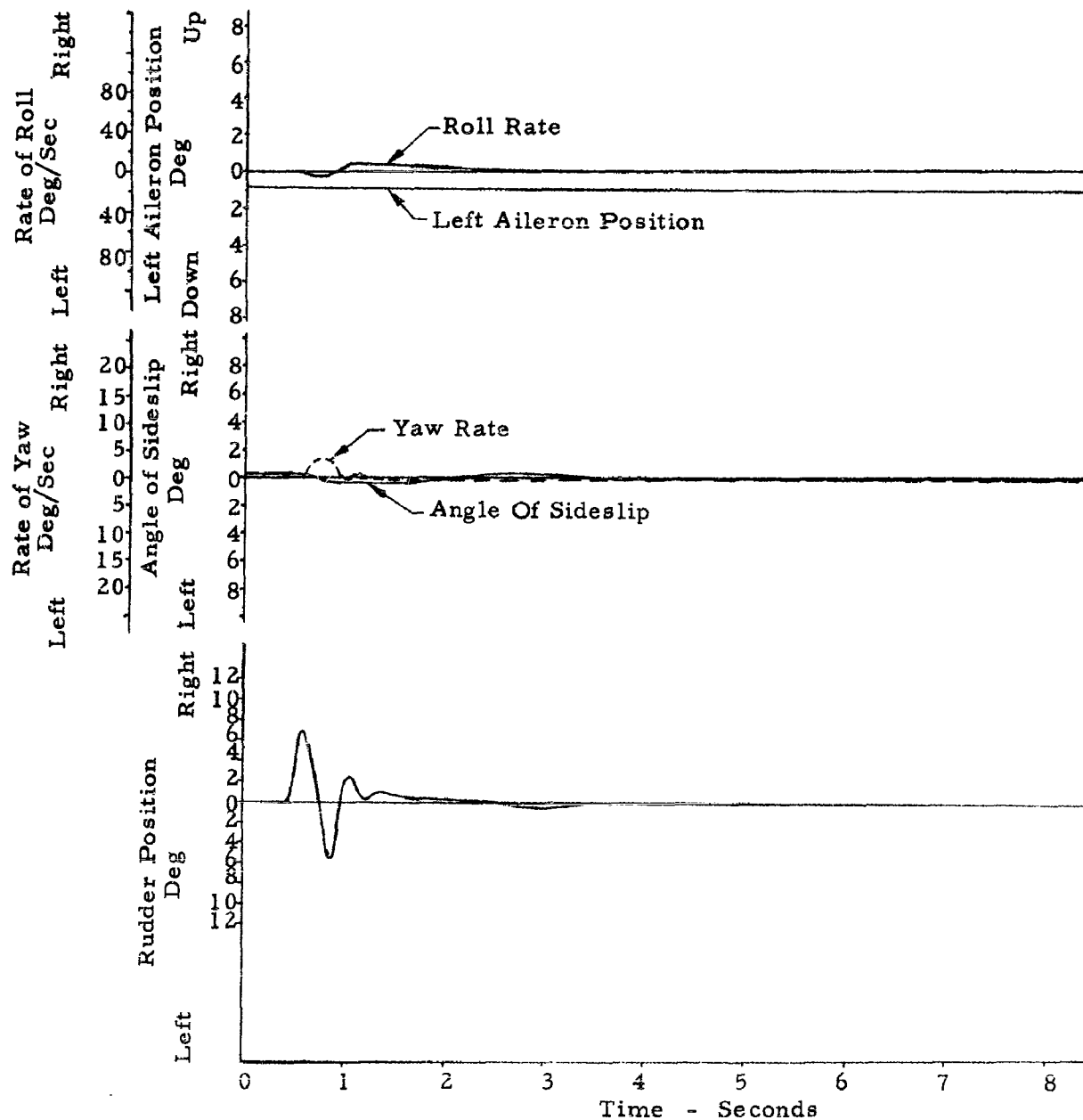


Figure No. 59
 DYNAMIC LATERAL-DIRECTIONAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers Off
 Altitude 35,050 ft.
 Mach No. .932

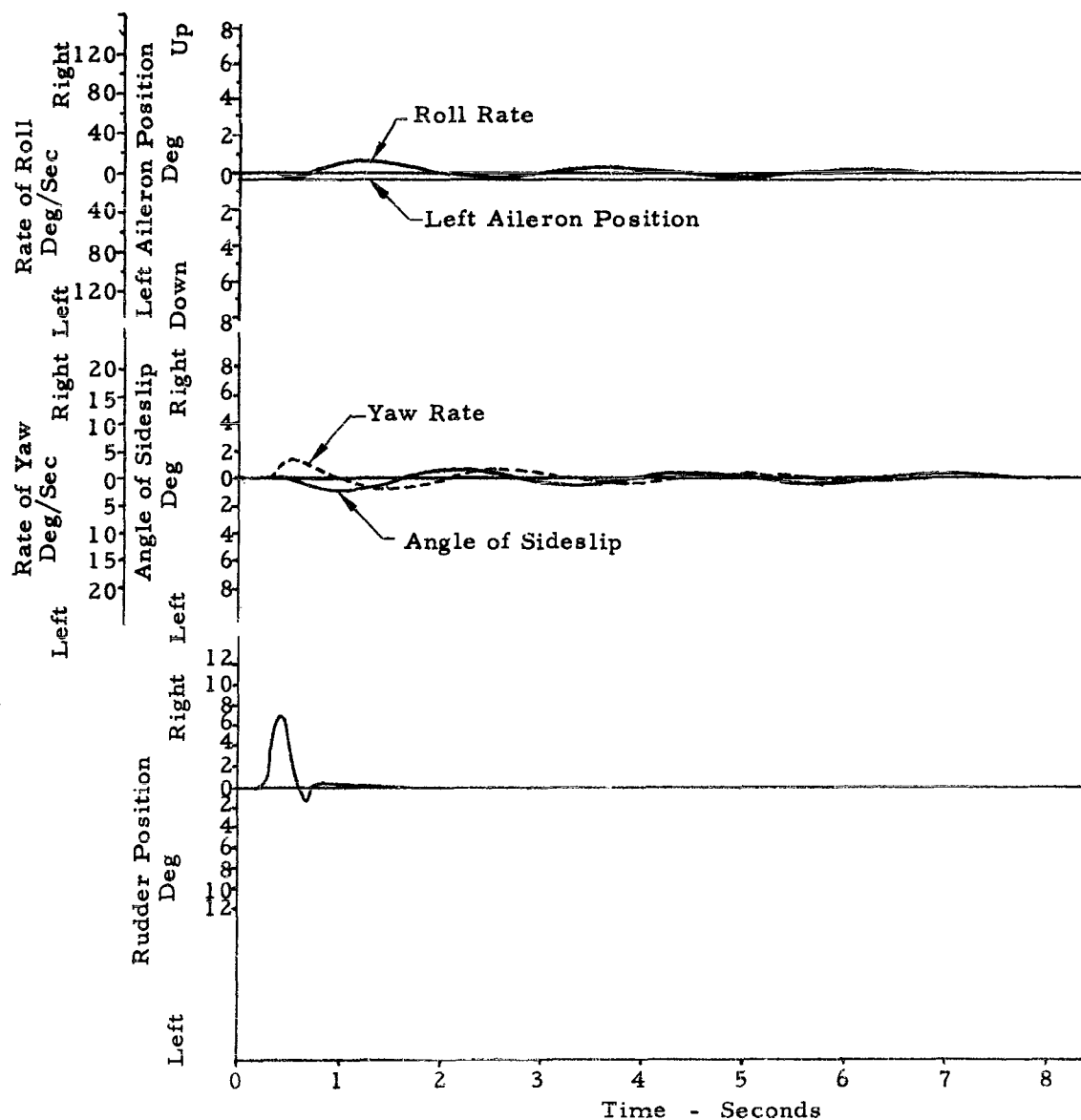


Figure No. 60
 DYNAMIC LATERAL-DIRECTIONAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers Off
 Altitude 34,840 ft.
 Mach No. 1.112

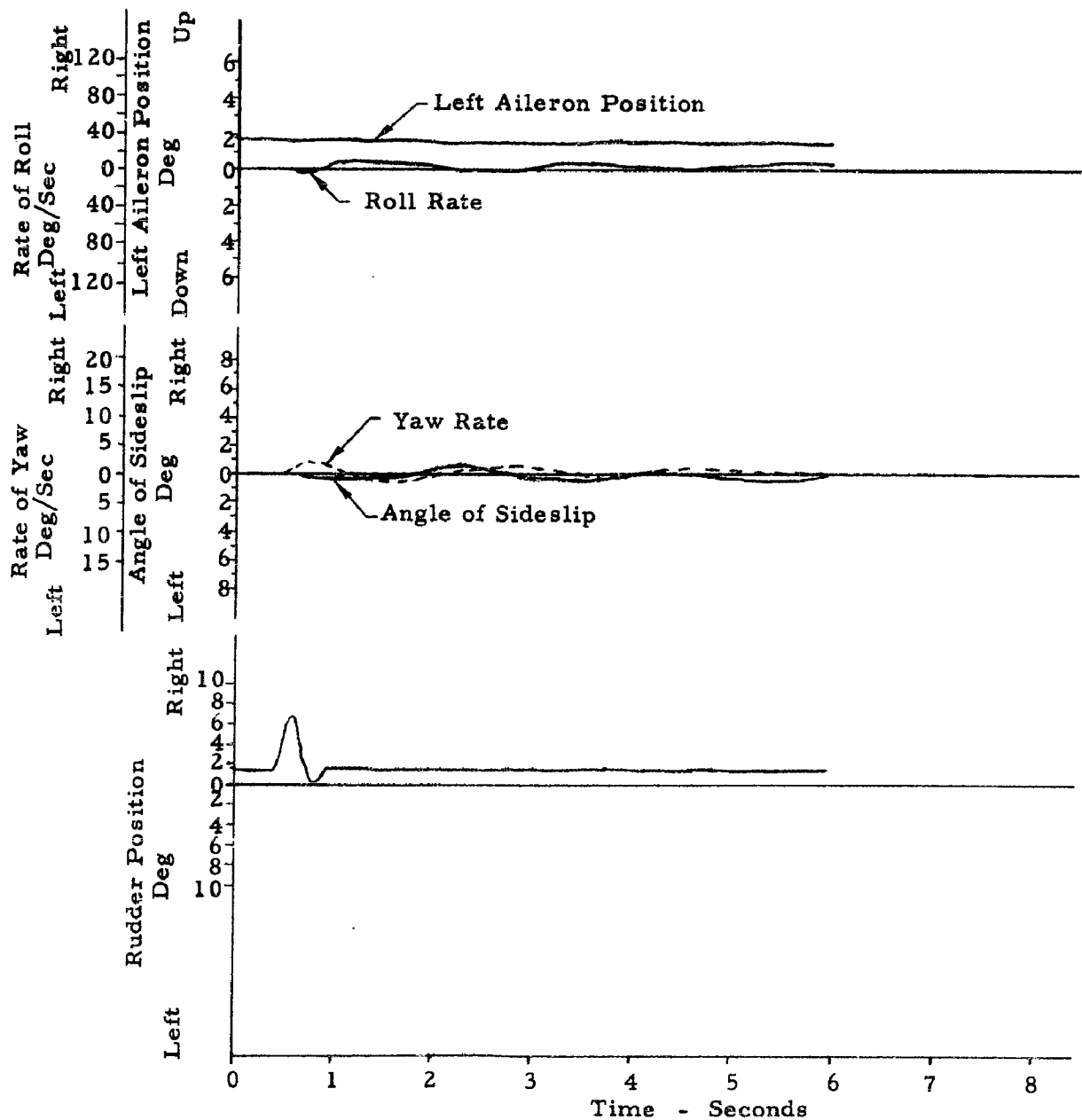


Figure No. 61
 DYNAMIC LATERAL-DIRECTIONAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 33,600 ft.
 Mach No. 1.180

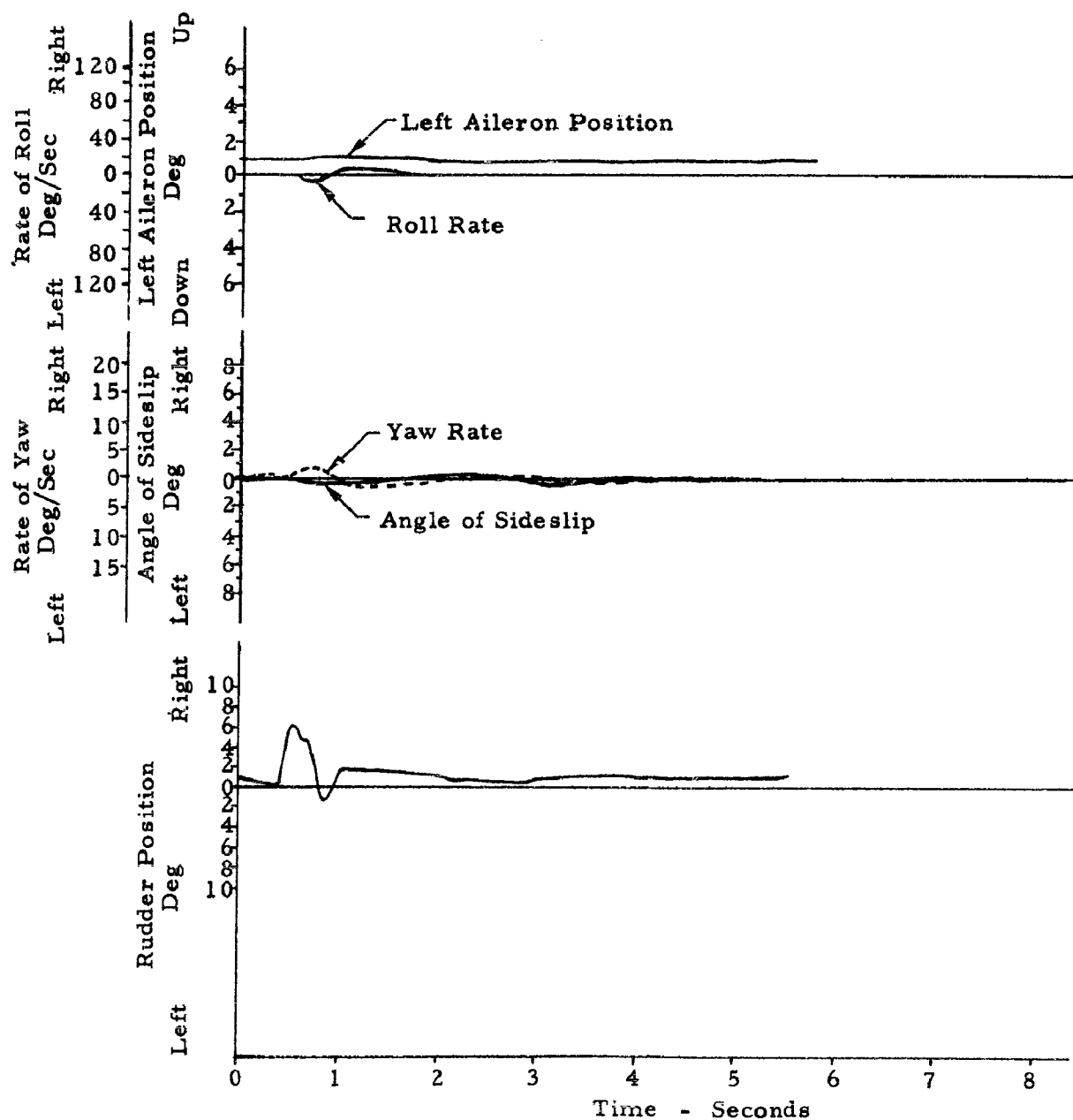


Figure No. 62
DYNAMIC LATERAL-DIRECTIONAL STABILITY
F-105B USAF No. 54-100
Cruise Configuration
Dampers Off
Altitude 34,690 ft.
Mach No. 1.413

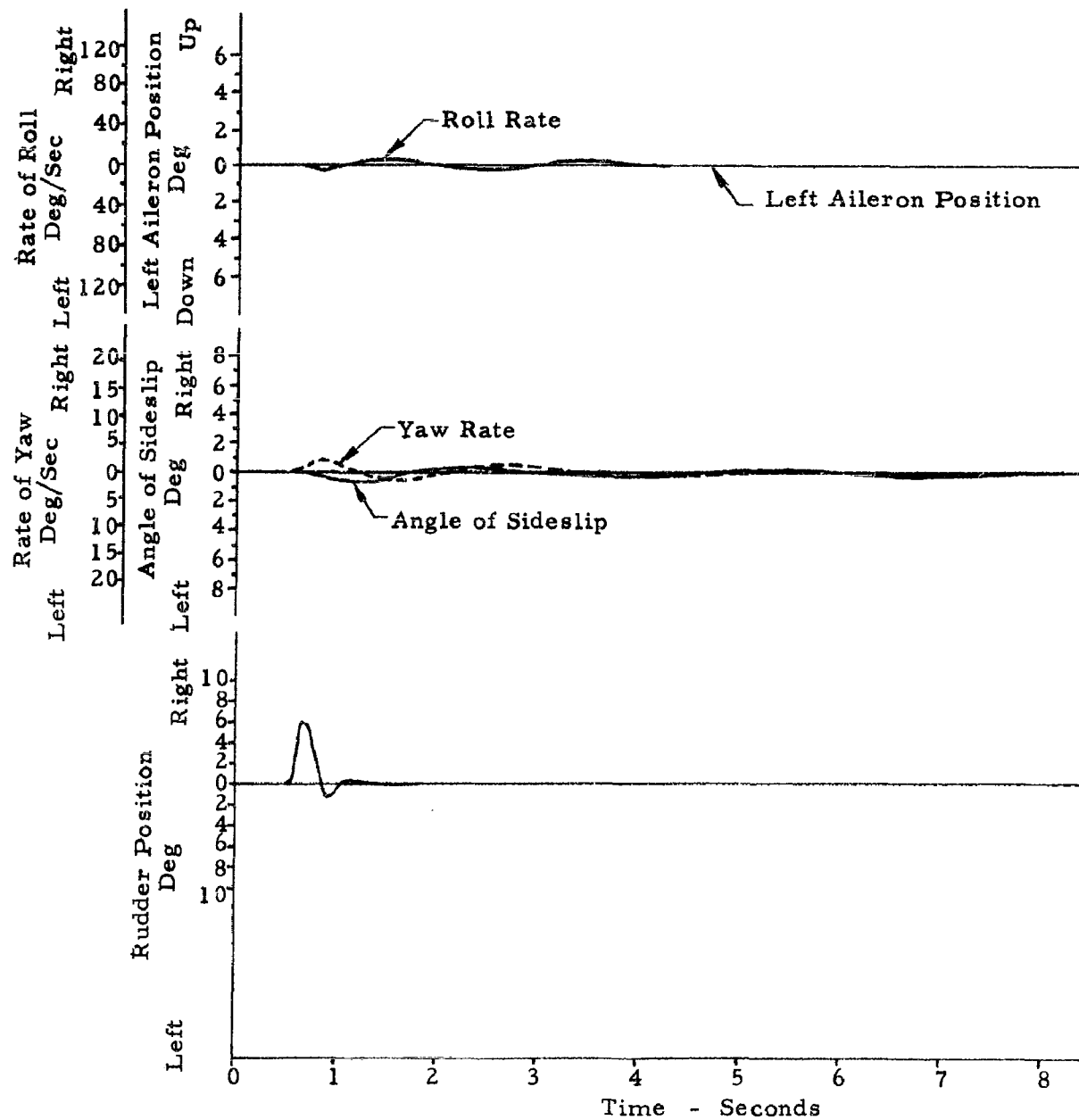


Figure No. 63
 DYNAMIC LATERAL-DIRECTIONAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 34,690 ft.
 Mach No. 1.413
 Pitch Damper Inoperative

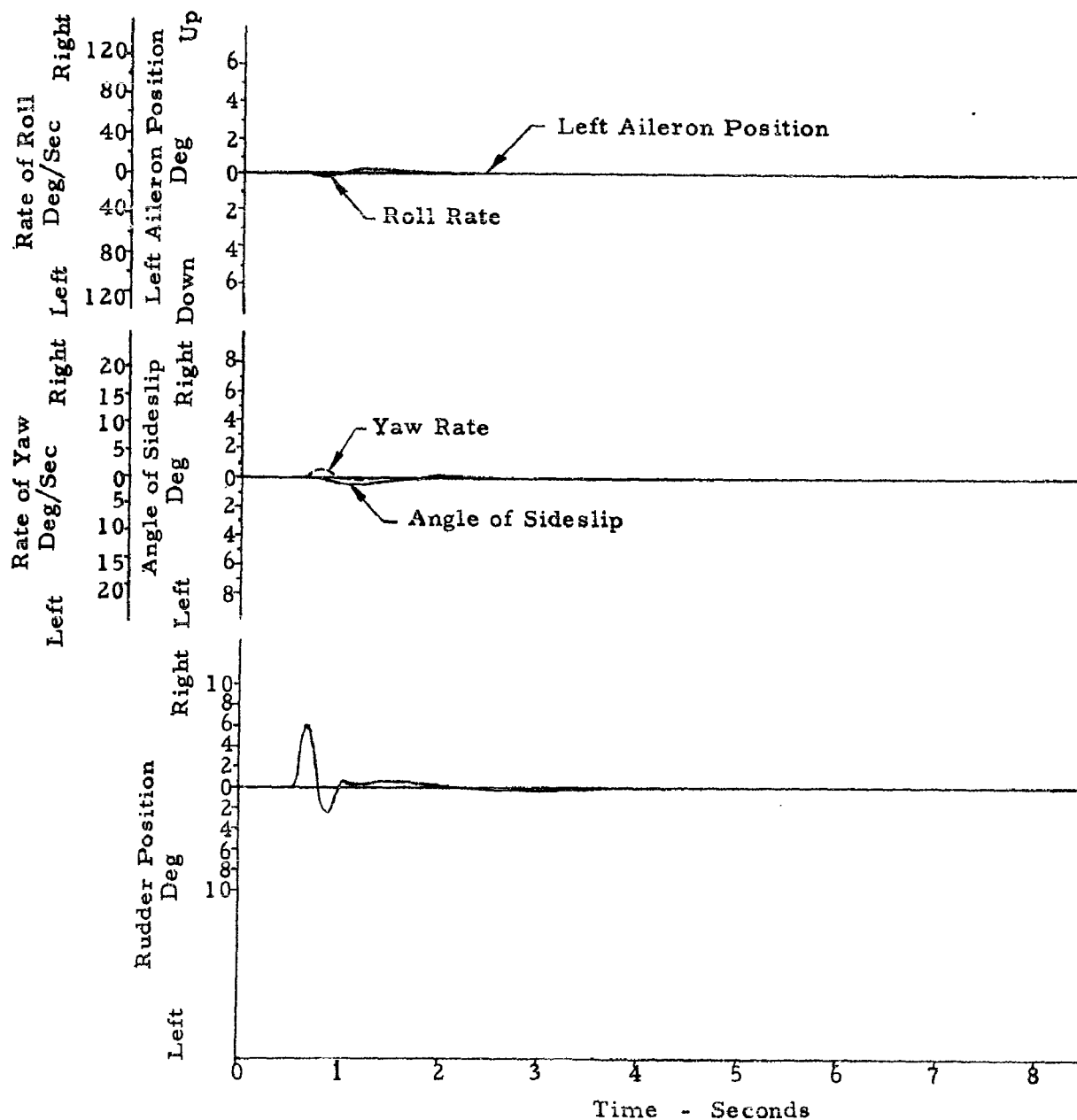


Figure No. 64
 DYNAMIC LATERAL-DIRECTIONAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 34,400 ft.
 Mach No. 1.70

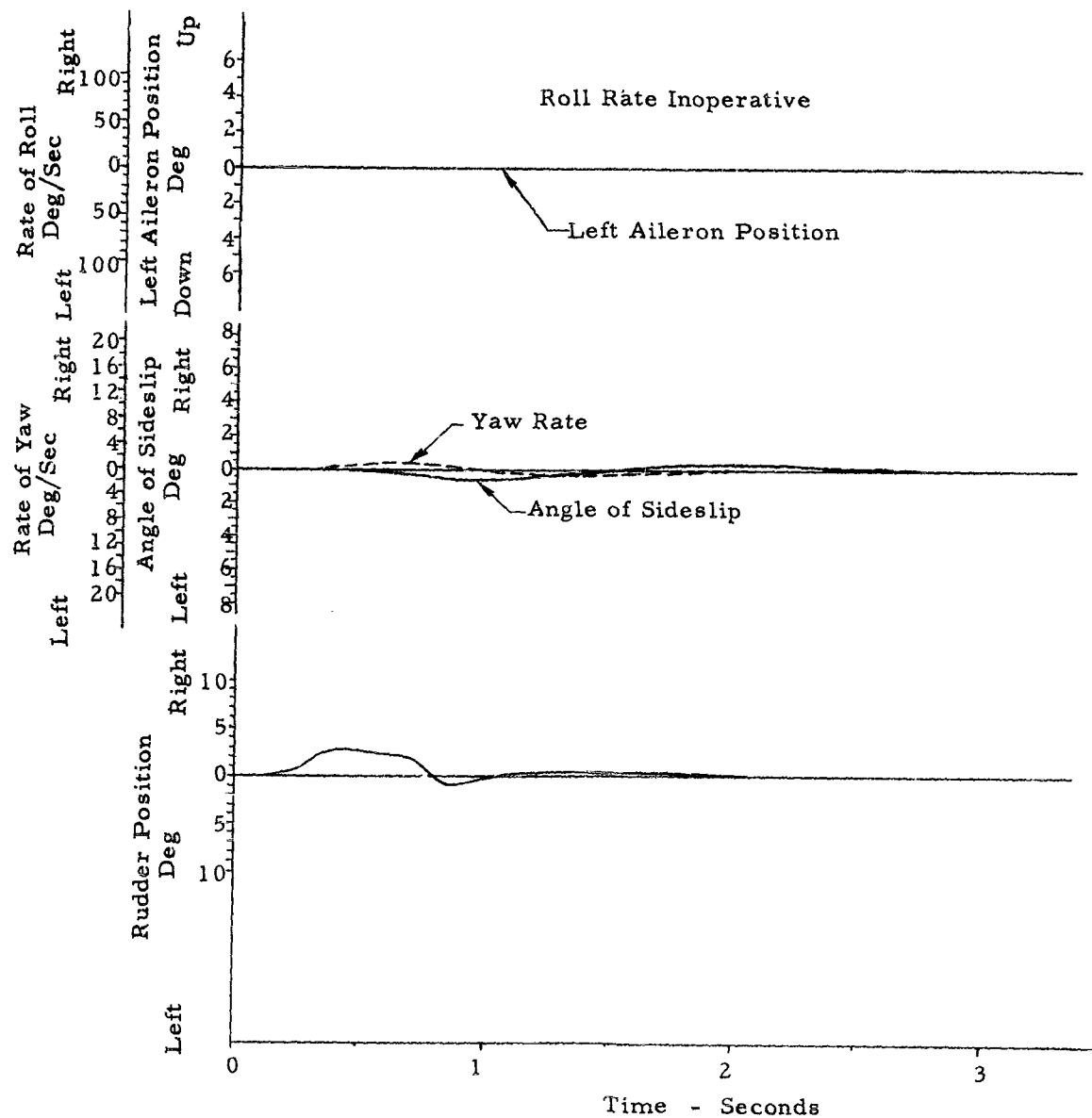


Figure No. 65
 DYNAMIC LATERAL-DIRECTIONAL STABILITY
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 44,700 ft.
 Mach No. 1.78

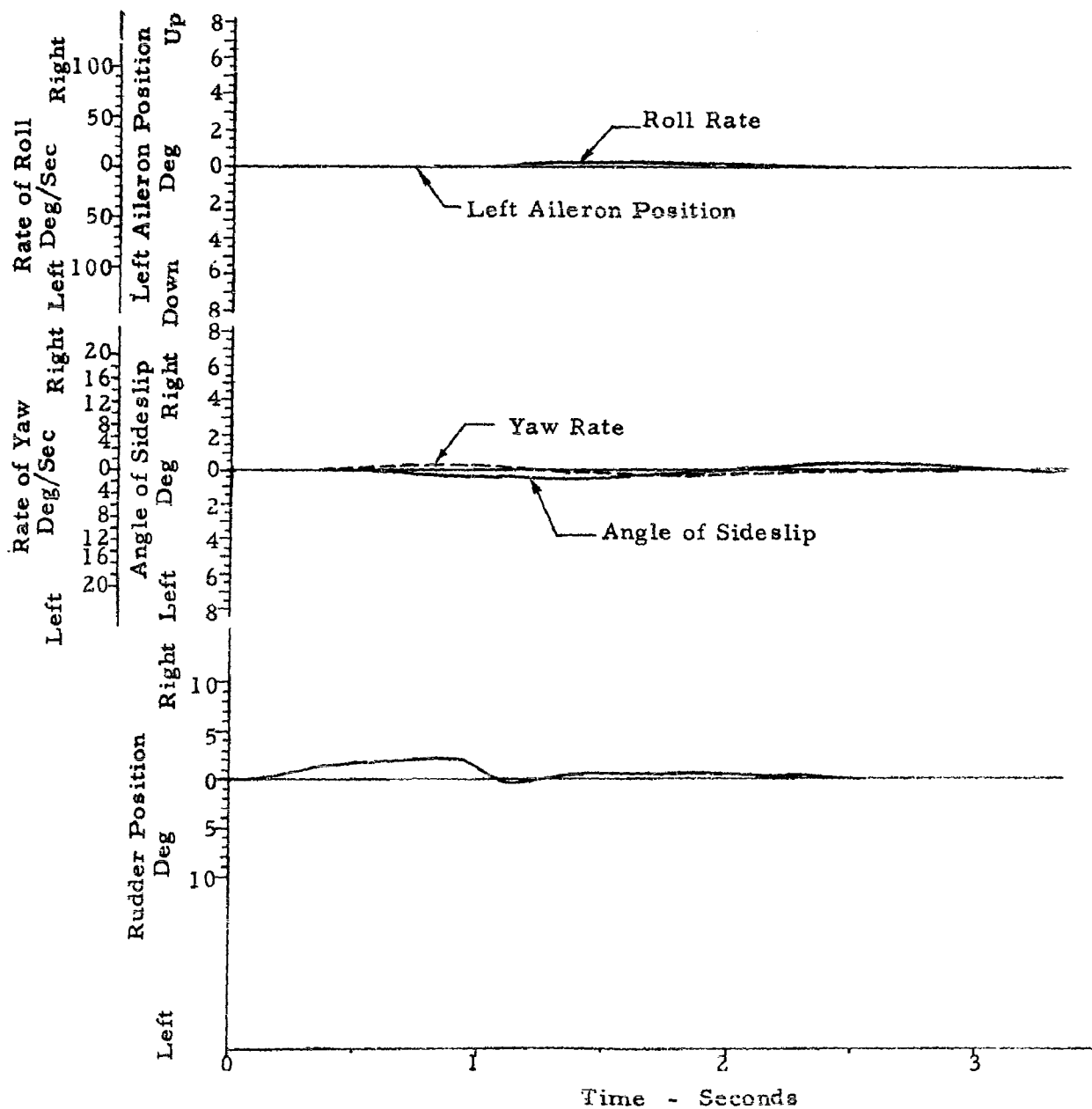


FIG. No. 66
ROLL SUMMARY
F-105B L19AF NO 54-100

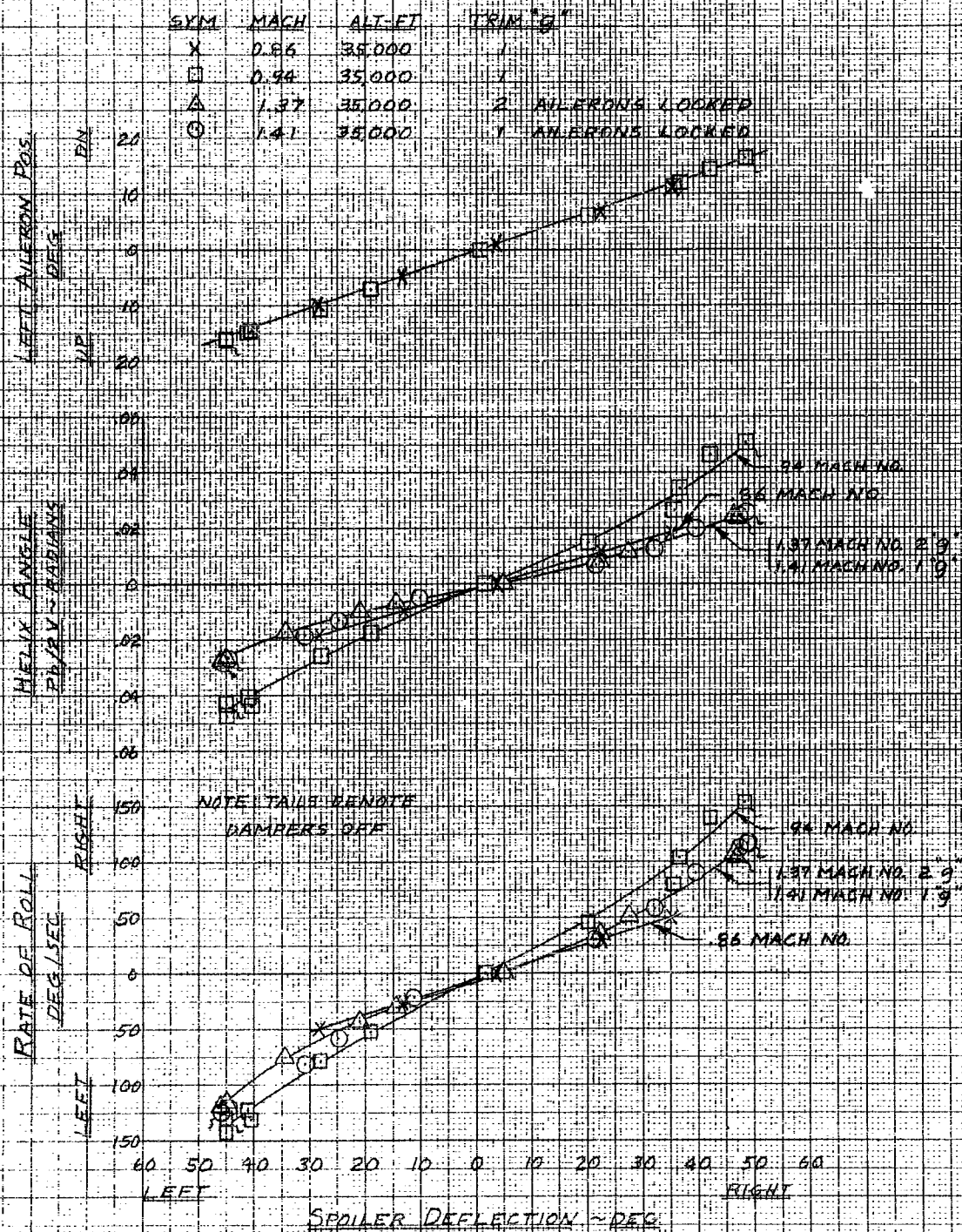


Fig. No. 47
ROLL SUMMARY
F-105B USAF NO. 54-102

<u>SYM</u>	<u>MACH</u>	<u>ALT - FT</u>	<u>TRIM %</u>	<u>AILERONS LOCKED</u>
0	1.40	45,000	1	

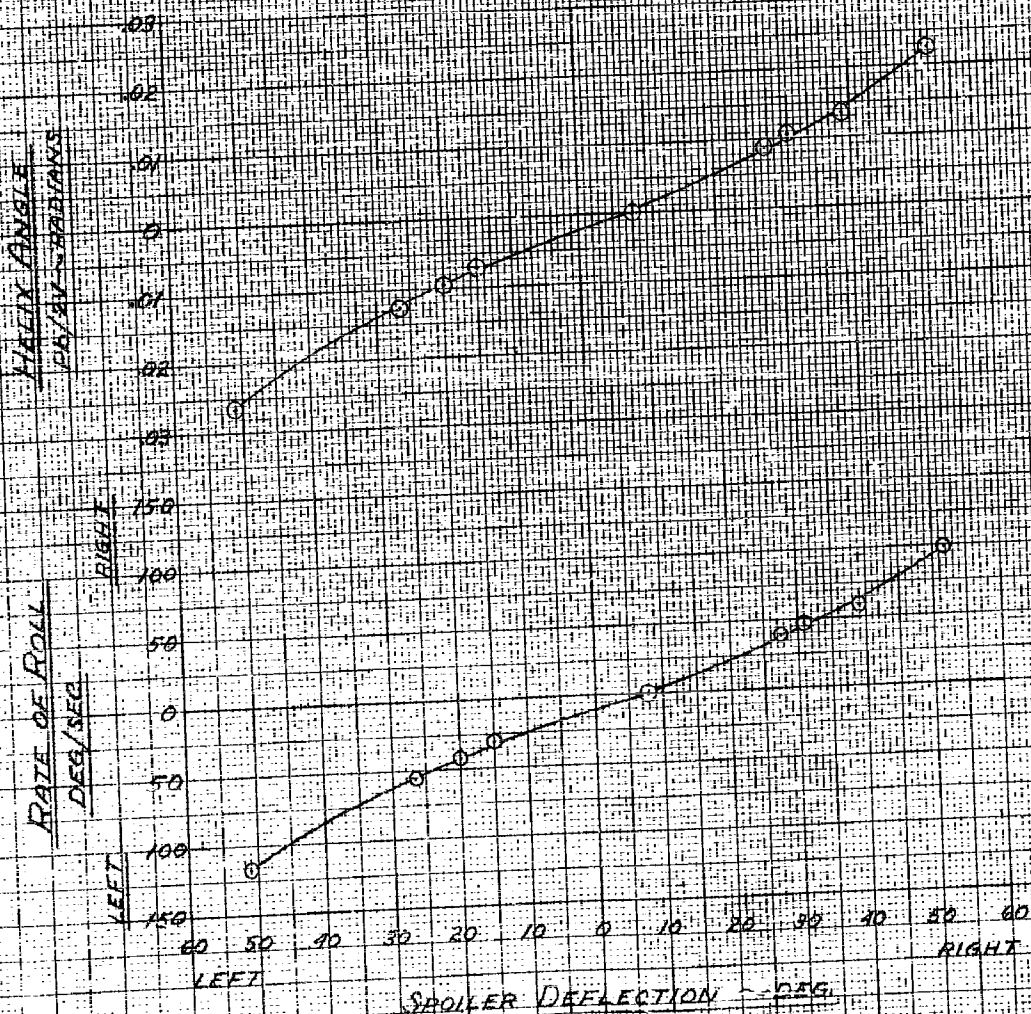


FIG. NO. 67
ROLL SUMMARY
F-105B USAF NO. 54-100

<u>SYM.</u>	<u>MACH</u>	<u>ALT. - FT.</u>	<u>TRIM °G</u>	<u>AILERONS LOCKED</u>
0	1.40	45,000	7	

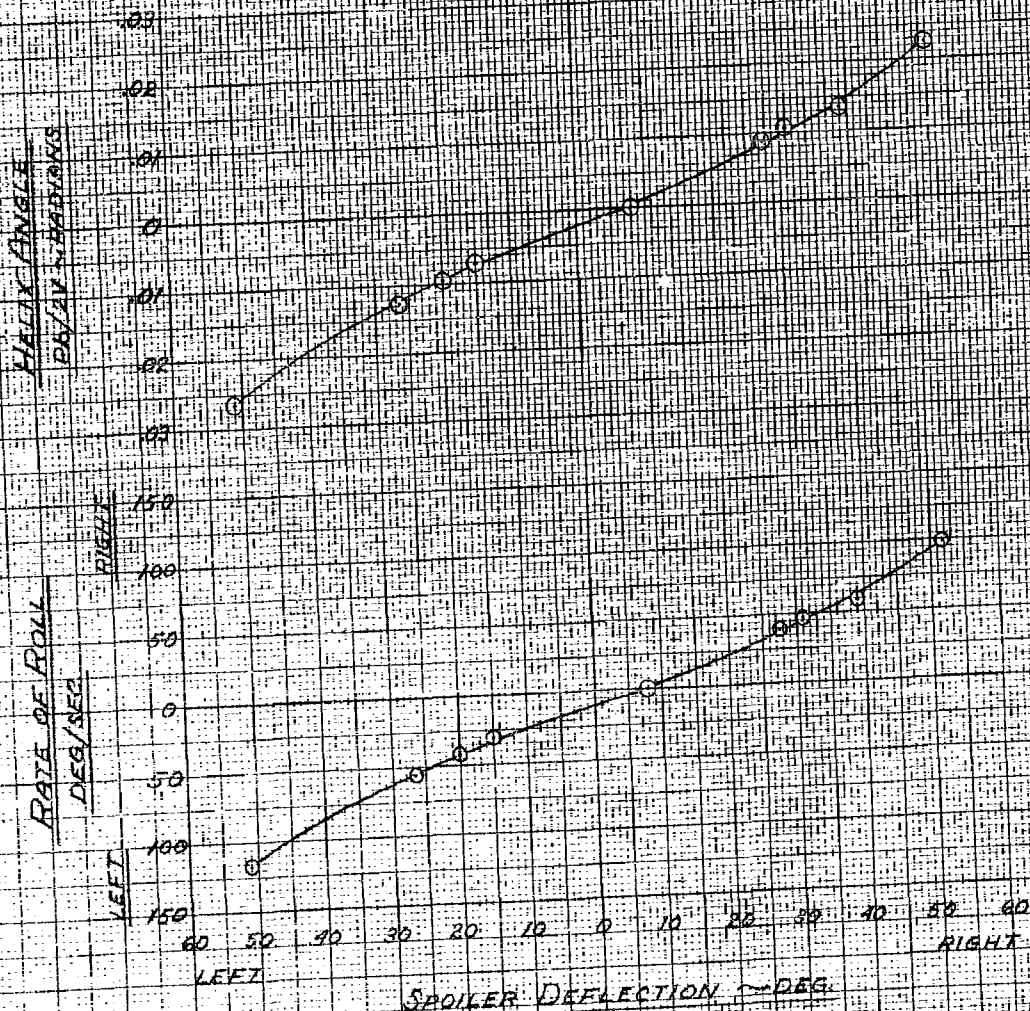


Figure No. 68
 AILERON ROLLS
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 35,700 ft.
 Mach No. .85

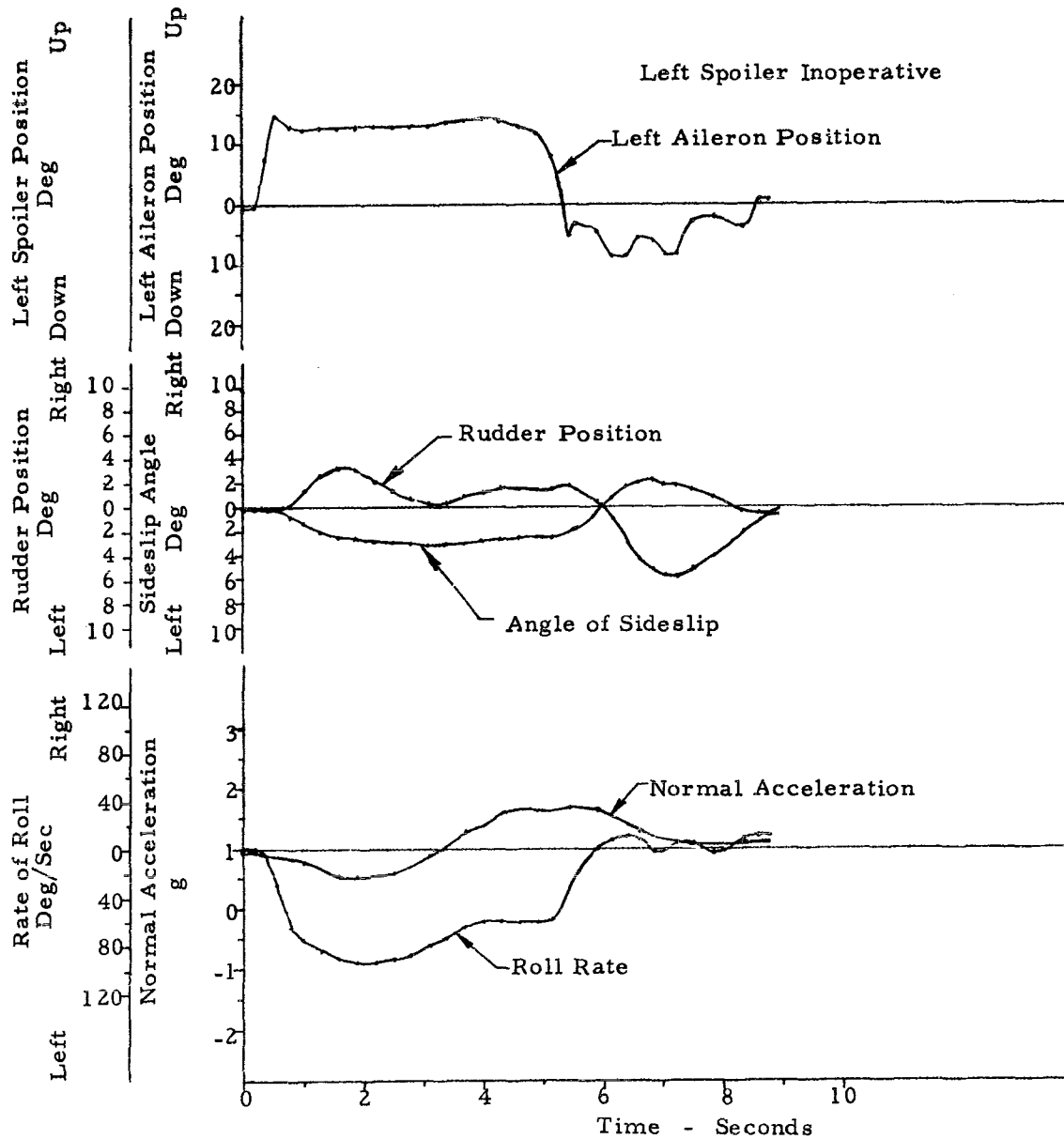


Figure No. 69
AILERON ROLLS
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers Off
 Altitude 33,600 ft.
 Mach No. .920

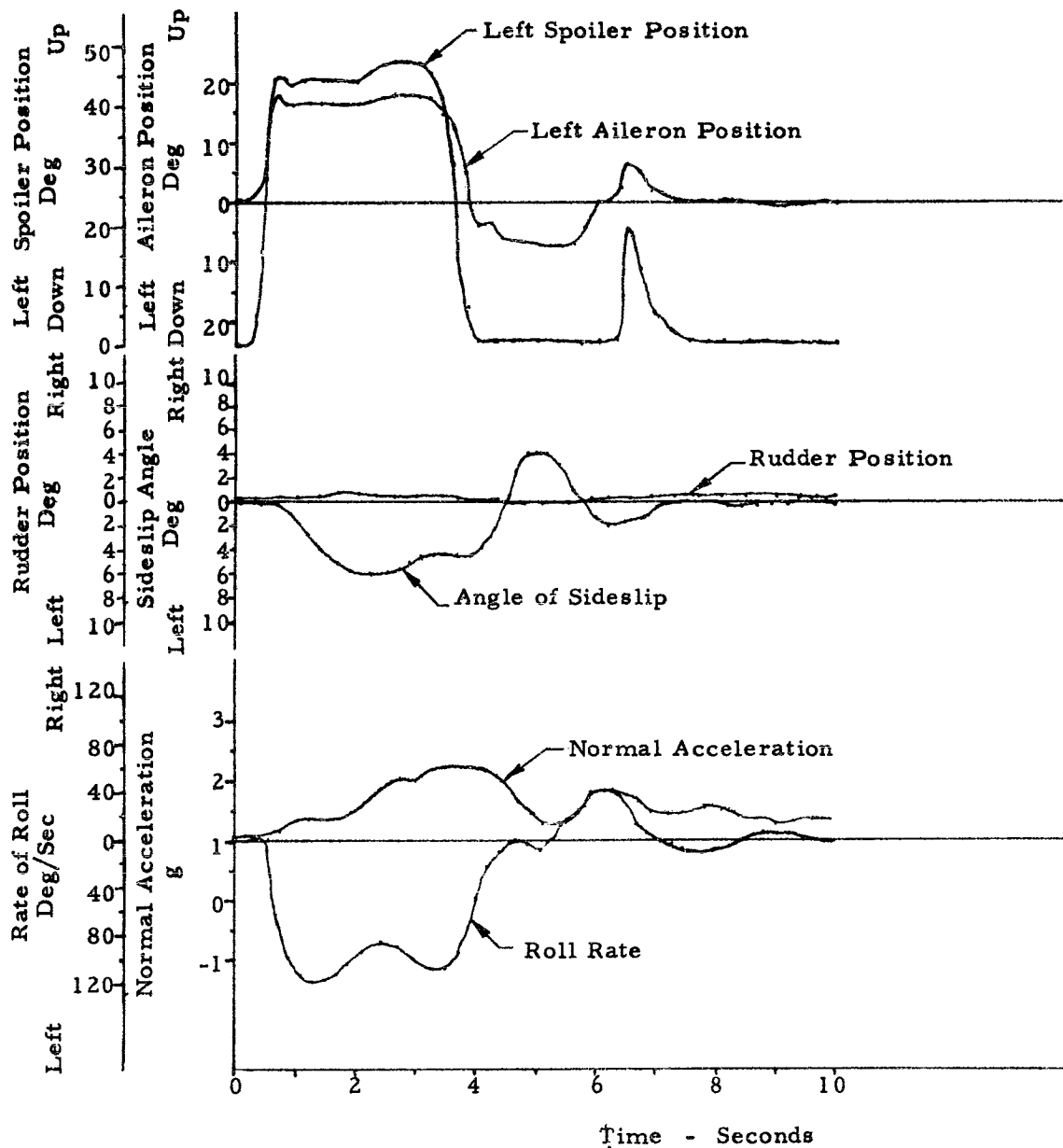


Figure No. 70
 AILERON ROLLS
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers Off
 Altitude 33,530 ft.
 Mach No. .910

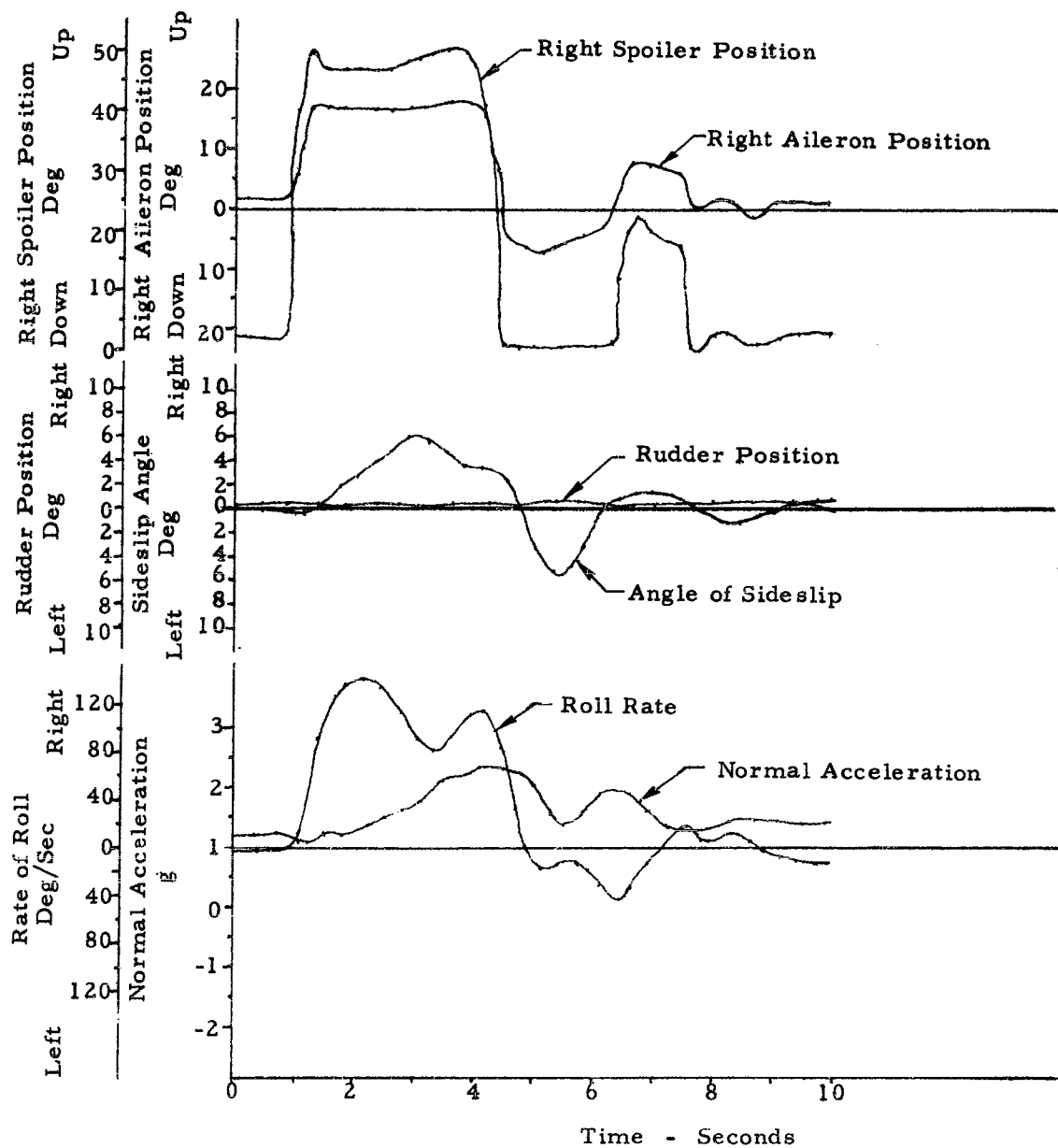


Figure No. 71
 AILERON ROLLS
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers Off
 Altitude 34,400 ft.
 Mach No. 1.41

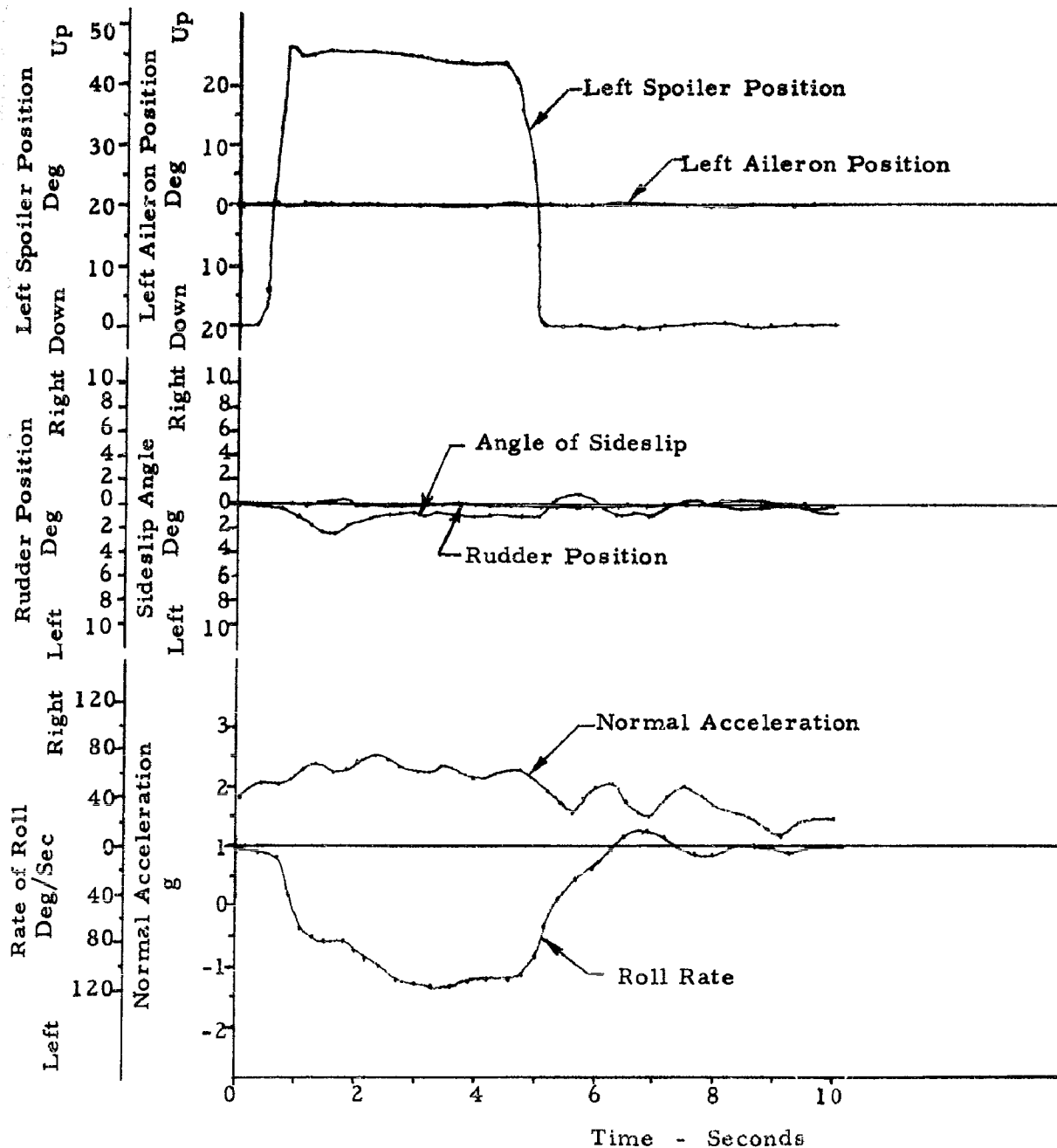


Figure No. 72
 AILERON ROLLS
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 34,700 ft.
 Mach No. 1.39

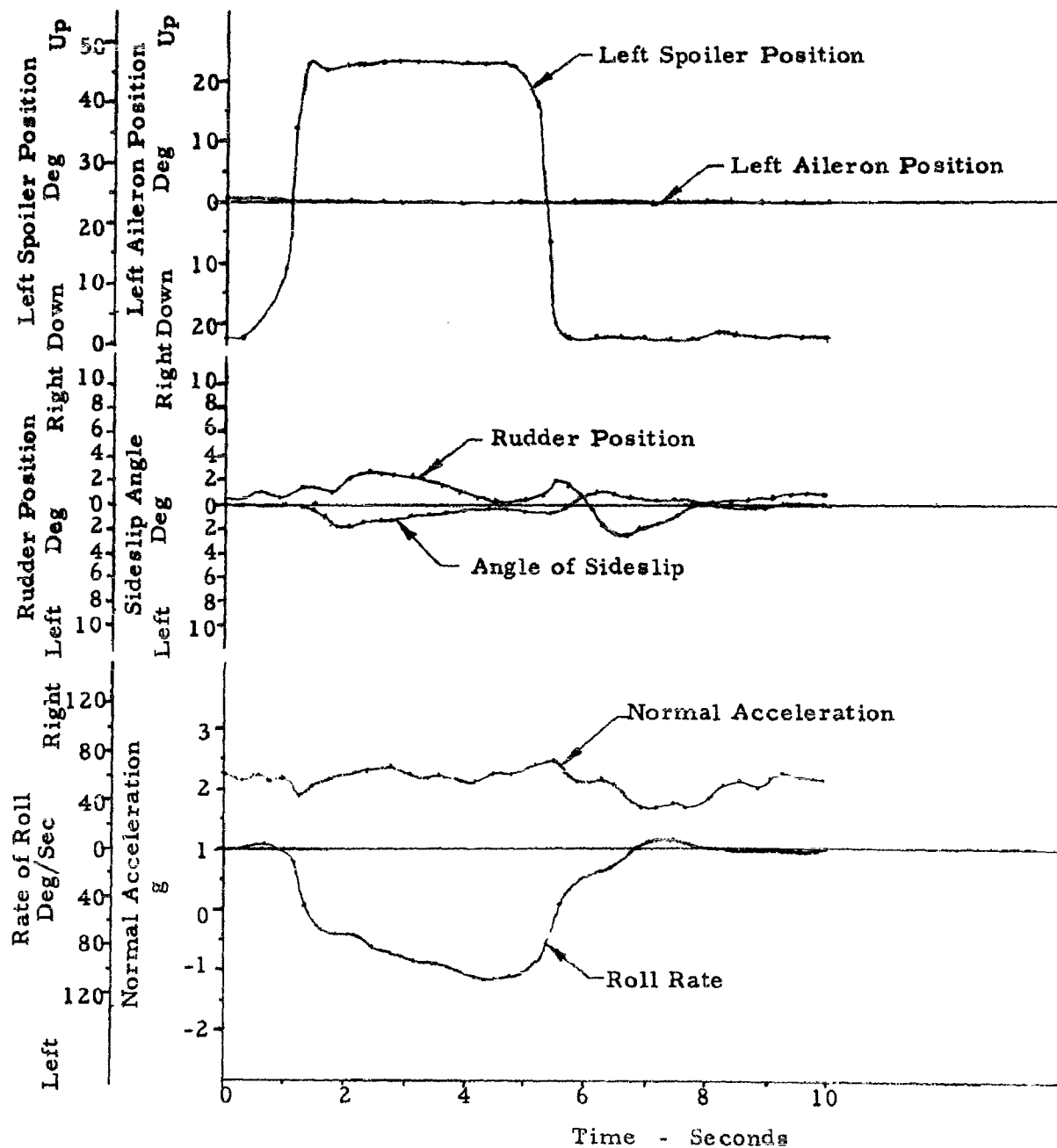


Figure No. 73
 AILERON ROLLS
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers Off
 Altitude 34,000 ft.
 Mach No. 1.47

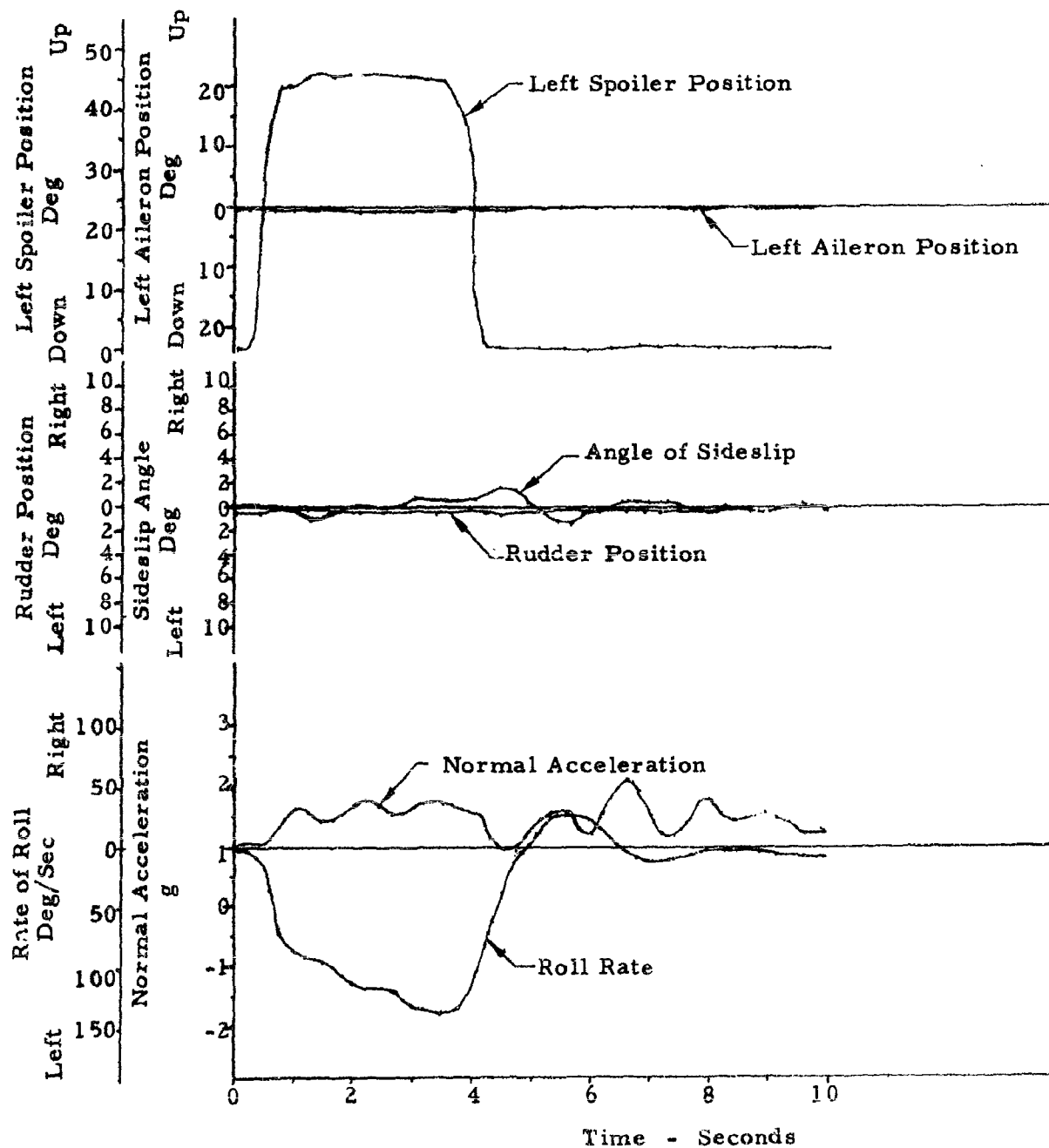


Figure No. 74
 AILERON ROLLS
 F-105B USAF No. 54-100
 Cruise Configuration
 Dampers On
 Altitude 42,700 ft.
 Mach No. 1.40

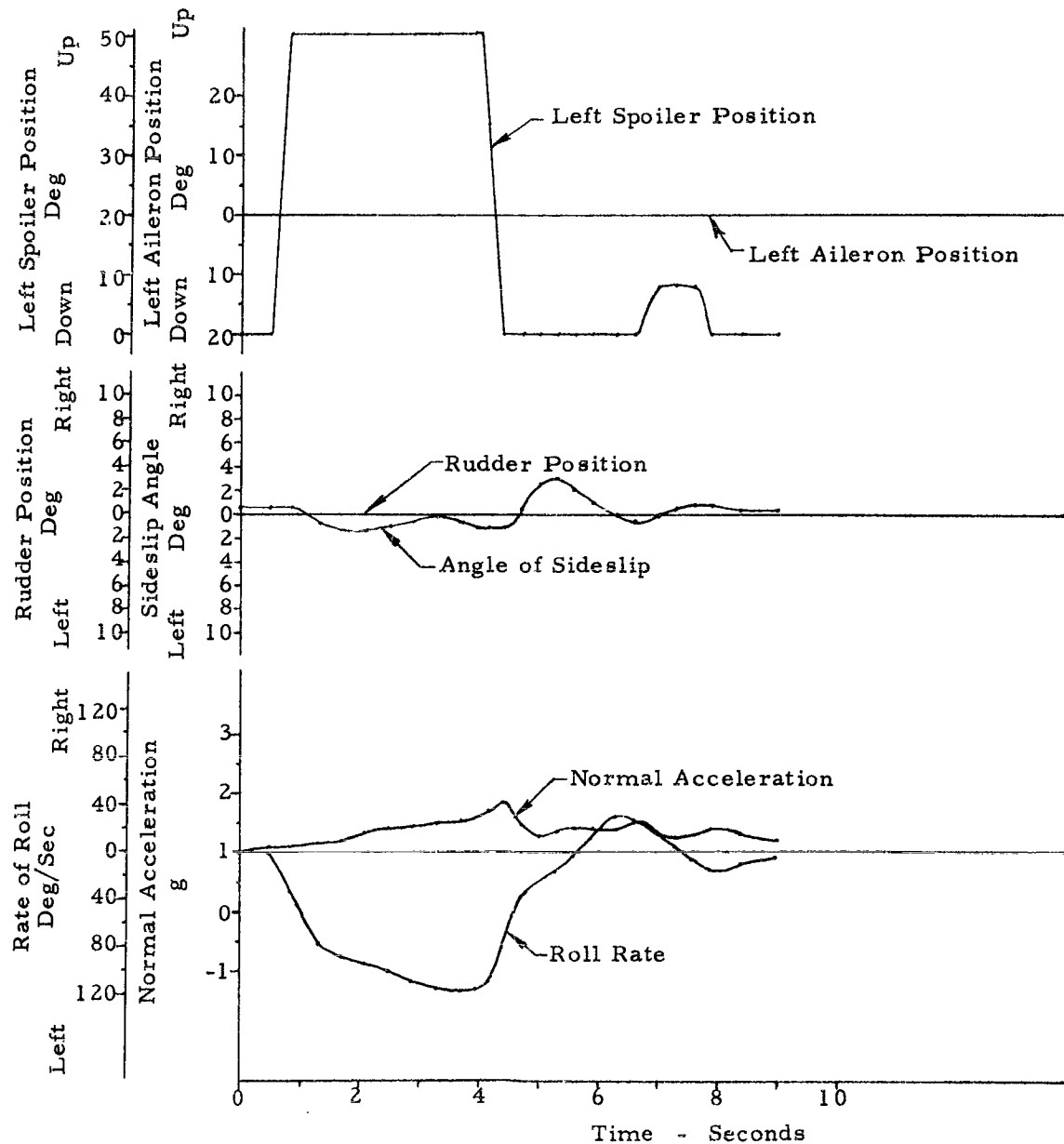


FIG NO 75

TIME HISTORY OF A SPEED BRAKE OPENING

F-105B USAF NO. 58-100

ALTITUDE 45,000 FT

MAXIMUM POWER

GROSS WEIGHT 30,700 LBS.

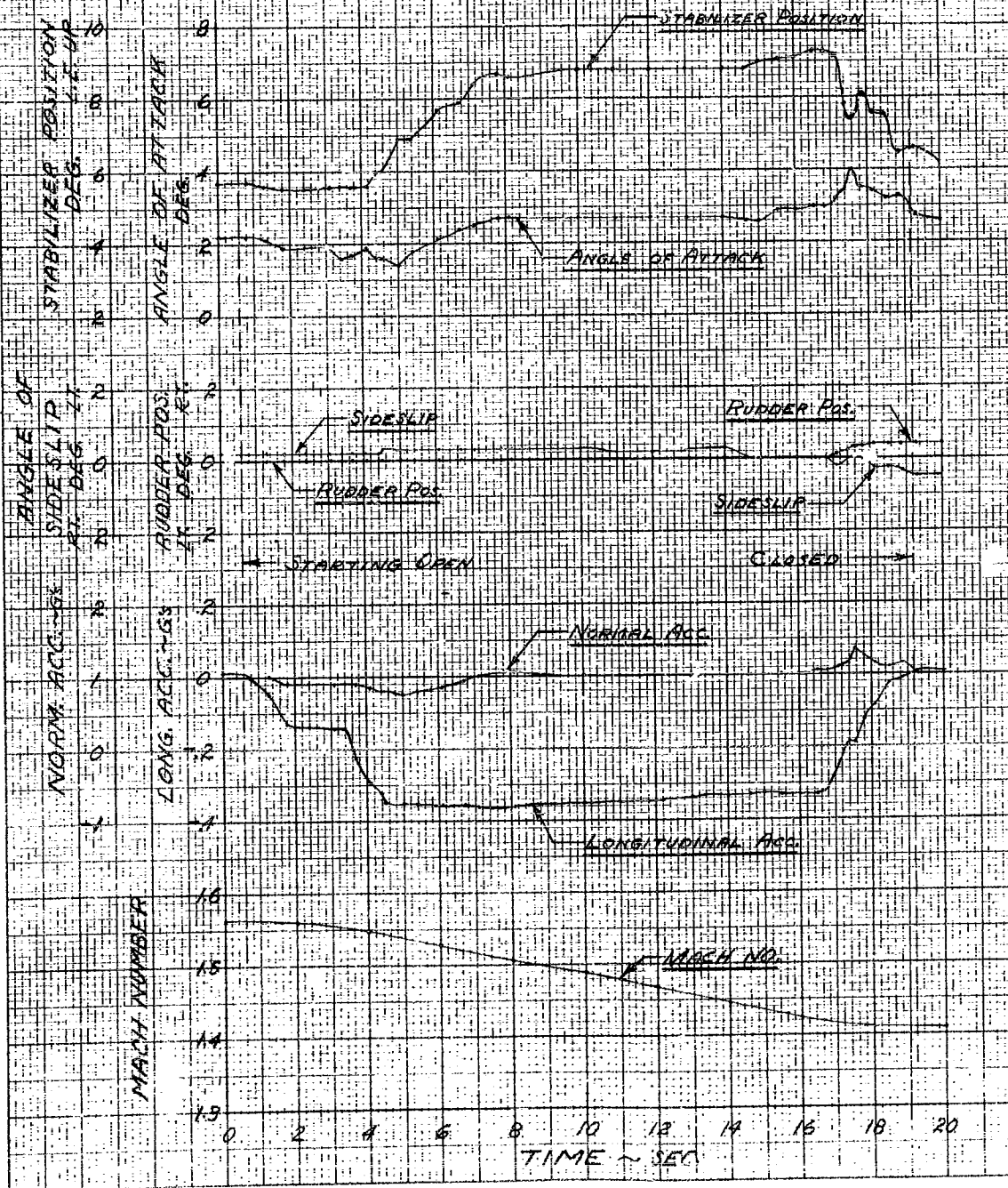


FIG. NO. 76
LONGITUDINAL CONTROL SYSTEM CALIBRATION
F-105B USAF NO. 54-100
28 FEB. 57
MECHANICAL ADVANTAGE RATIO 1:1

NOTE: EACH POINT REPRESENTS THE
STICK FORCE AT WHICH THE
CONTROL SURFACE BEGINS TO
MOVE WHEN THE STICK FORCE
IS SMOOTHLY INCREASED OR
DECREASED FROM A GIVEN
VALUE

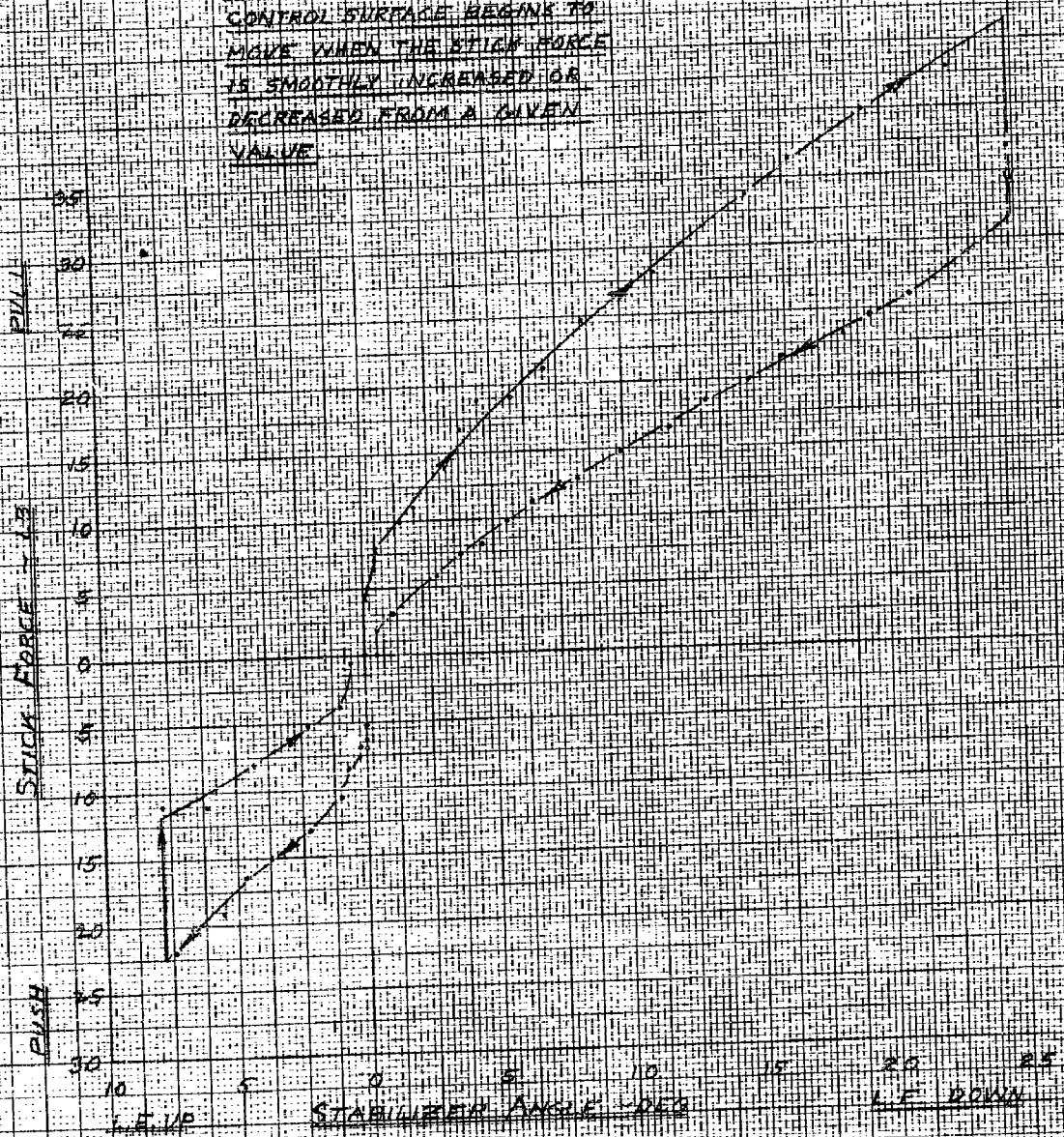


FIG. NO. 11
LONGITUDINAL CONTROL SYSTEM CALIBRATION
F-105B USAF NO. 58-100
7 JAN 57
MECHANICAL ADVANTAGE RATIO 1.1

NOTE: EACH POINT REPRESENTS THE
FINAL STABILIZED POSITION OF
THE CONTROL SURFACE AFTER
APPROACHING AND HOLDING A
CONSTANT STICK FORCE WITH
A HUNTER SPRING SCALE.

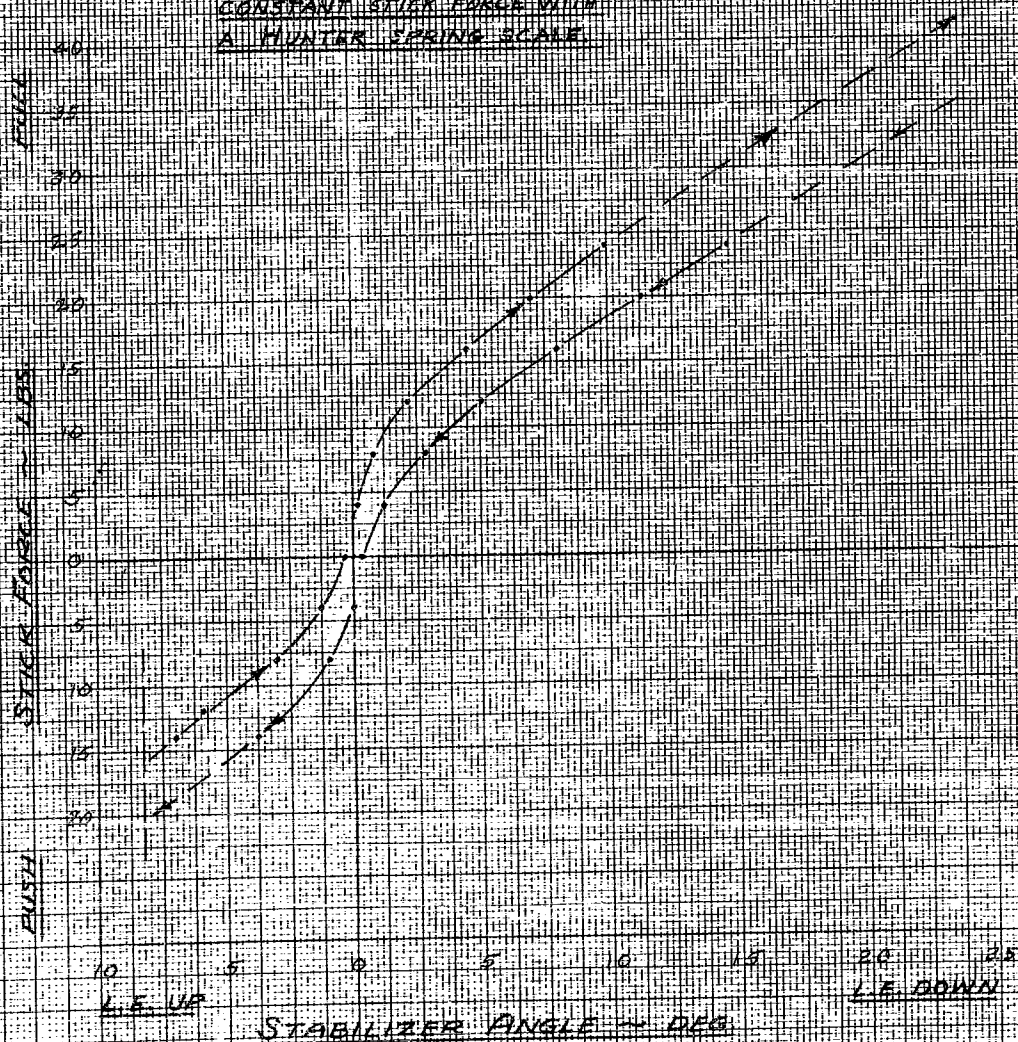


FIG. NO. 76
LONGITUDINAL CONTROL SYSTEM CALIBRATION
F-105B USAF NO. 58-100
9 JAN 57
MECHANICAL ADVANTAGE RATIO 3.11

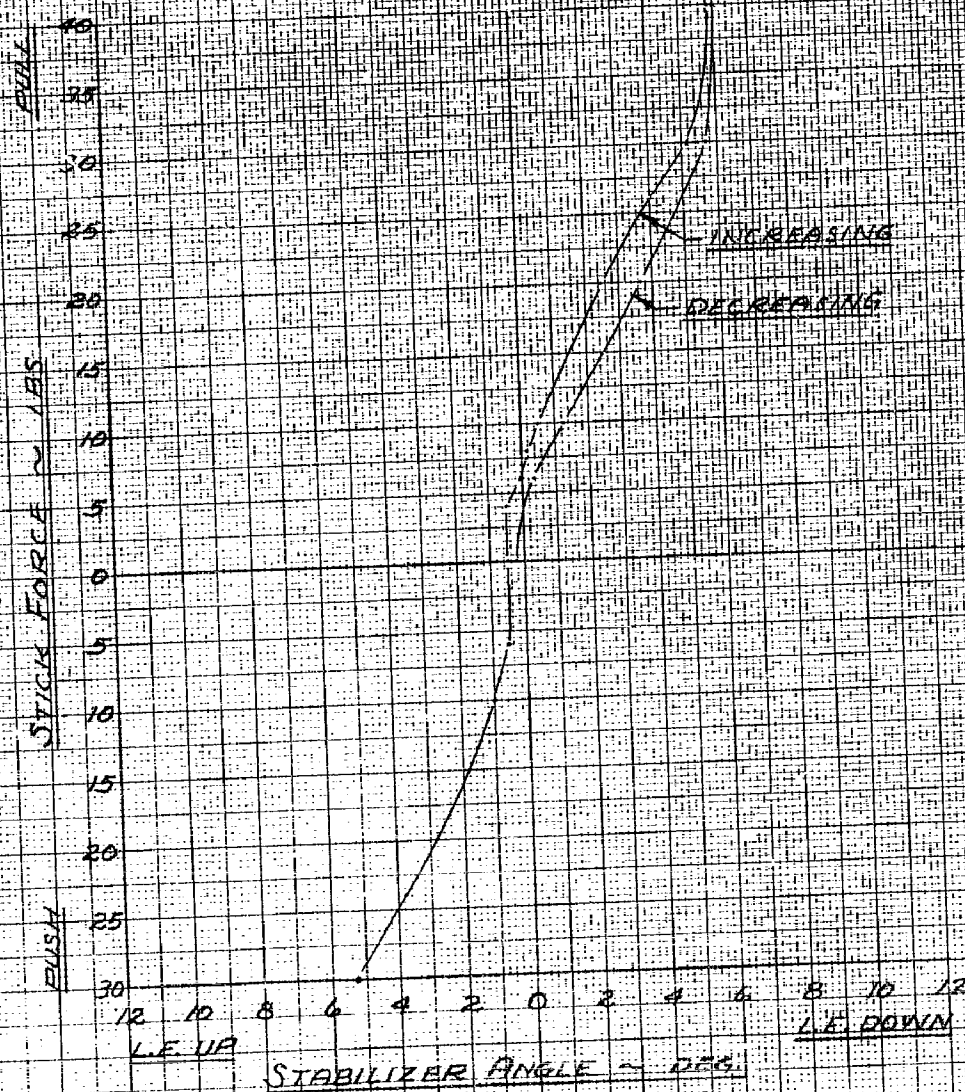


FIG. NO. 79
LATERAL CONTROL SYSTEM CALIBRATION
AILERONS
F-105B, VERB NO. 54-100
12 JAN. 57

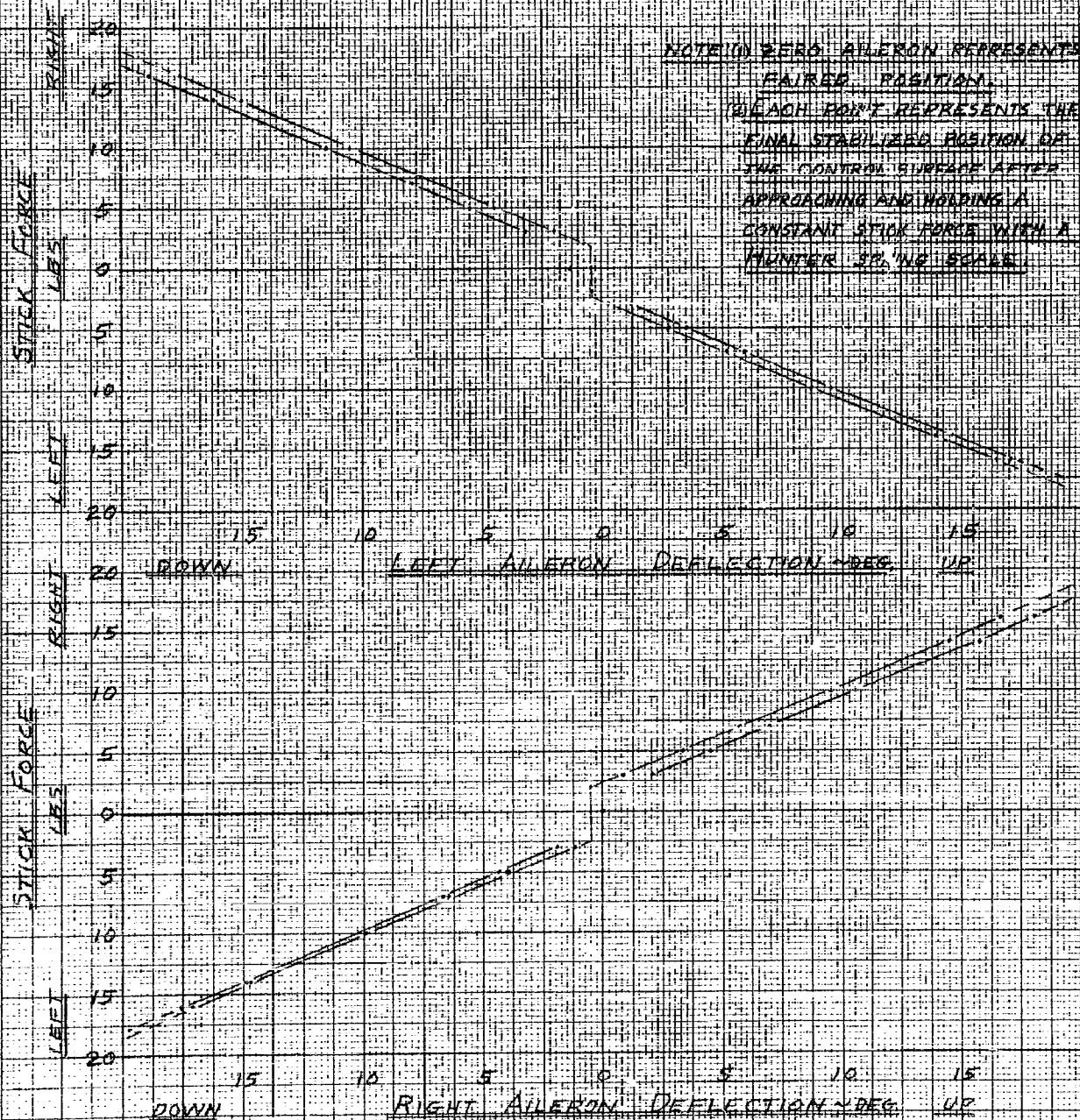


FIG. NO. 50
LATERAL CONTROL SYSTEM CALIBRATION
SPOILERS
 F-105B WAF NO. 54-1100
 12 MAY 57

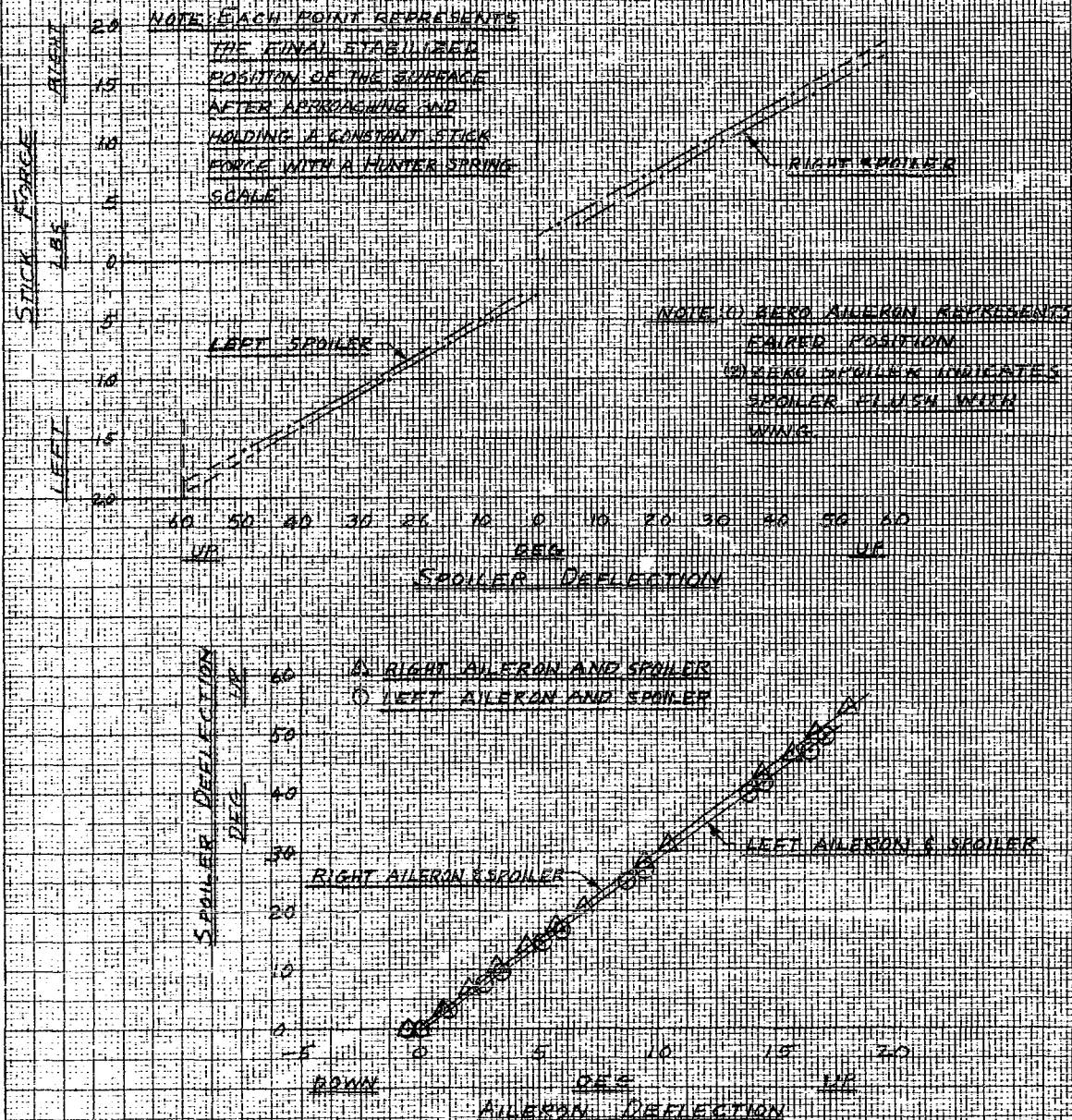


FIG. NO. 81
LATERAL CONTROL SYSTEM CALIBRATION
AILERONS

F-105B USAF NO. 54-100
 28 FEB 67

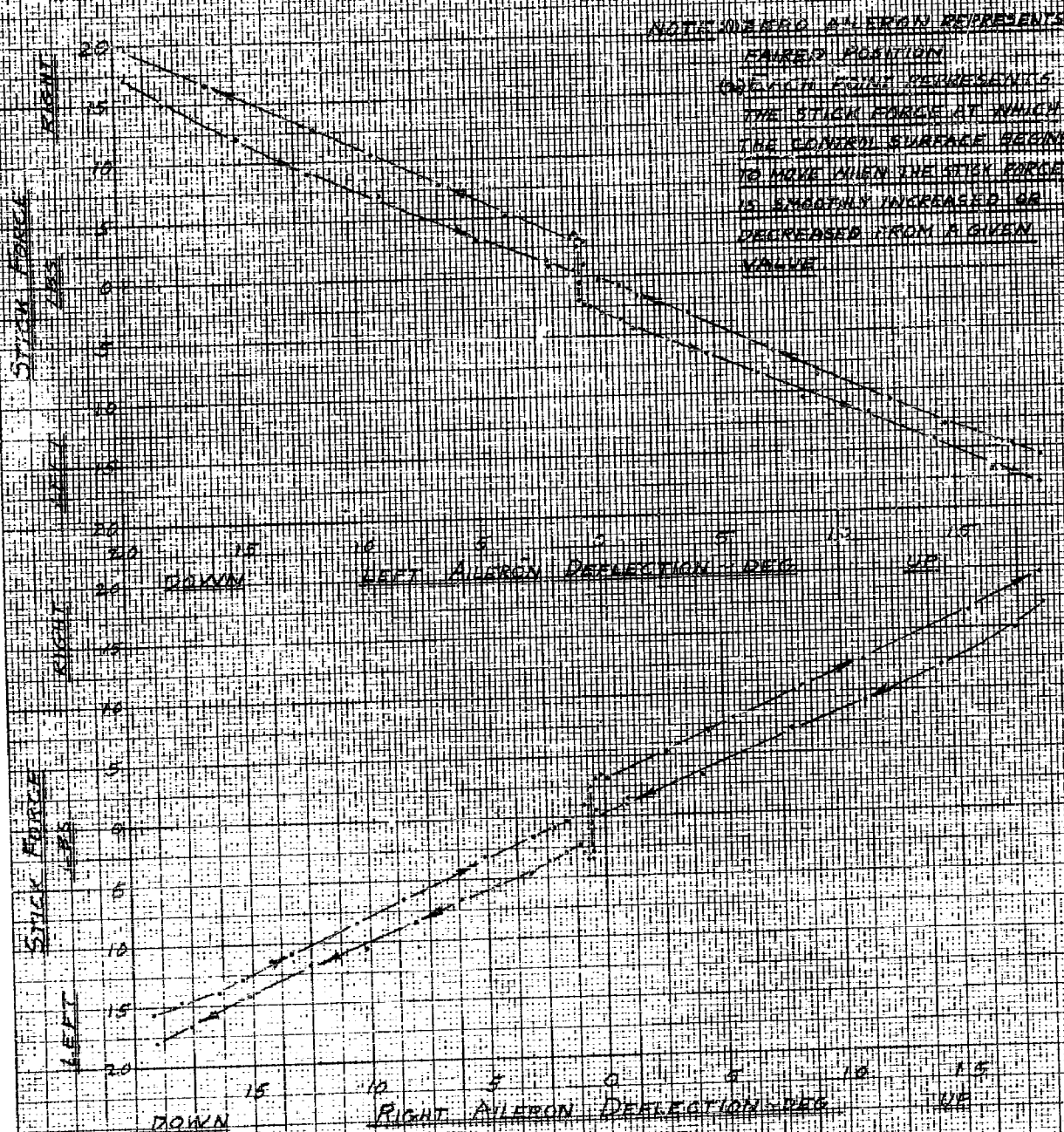


FIG. NO. 82
 LATERAL CONTROL SYSTEM CALIBRATION
 SPOILERS
 F-105B USAF NO. 54-100
 28 FEB 57

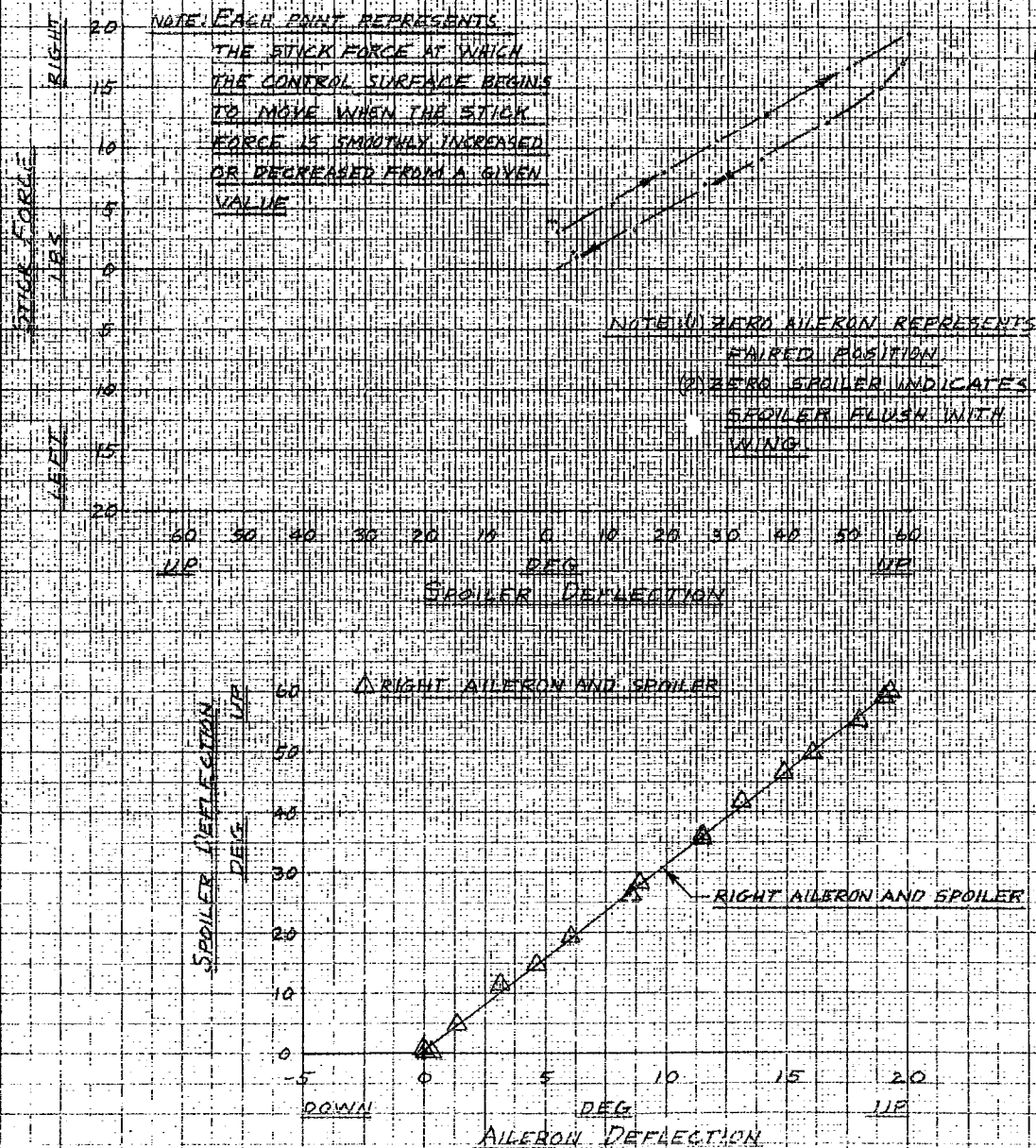


FIG. NO. 83
RUDDER CONTROL SYSTEM CALIBRATION
F-105B USAF NO. 54-100
12 JAN. 57
MECHANICAL ADVANTAGE RATIO 1:1
(BELOW 210 KNOTS INDICATED AIRSPEED)

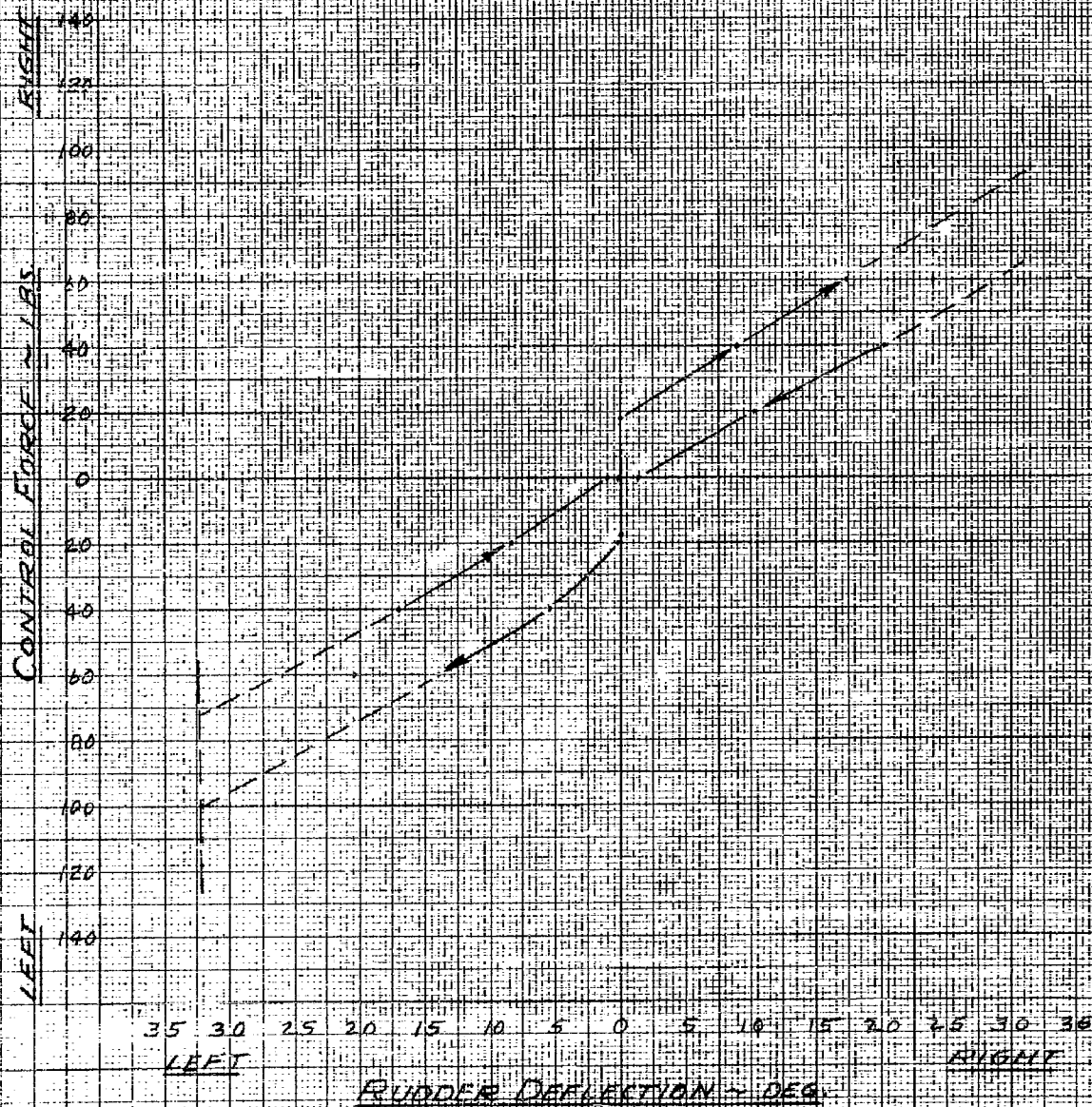


Fig No 84
RUDER CONTROL SYSTEM CALIBRATION
F105B USAF NO 54-100
12 JAN 57
MECHANICAL ADVANTAGE RATIO 4:1
(ABOVE 210 KNOTS INDICATED AIRSPEED)

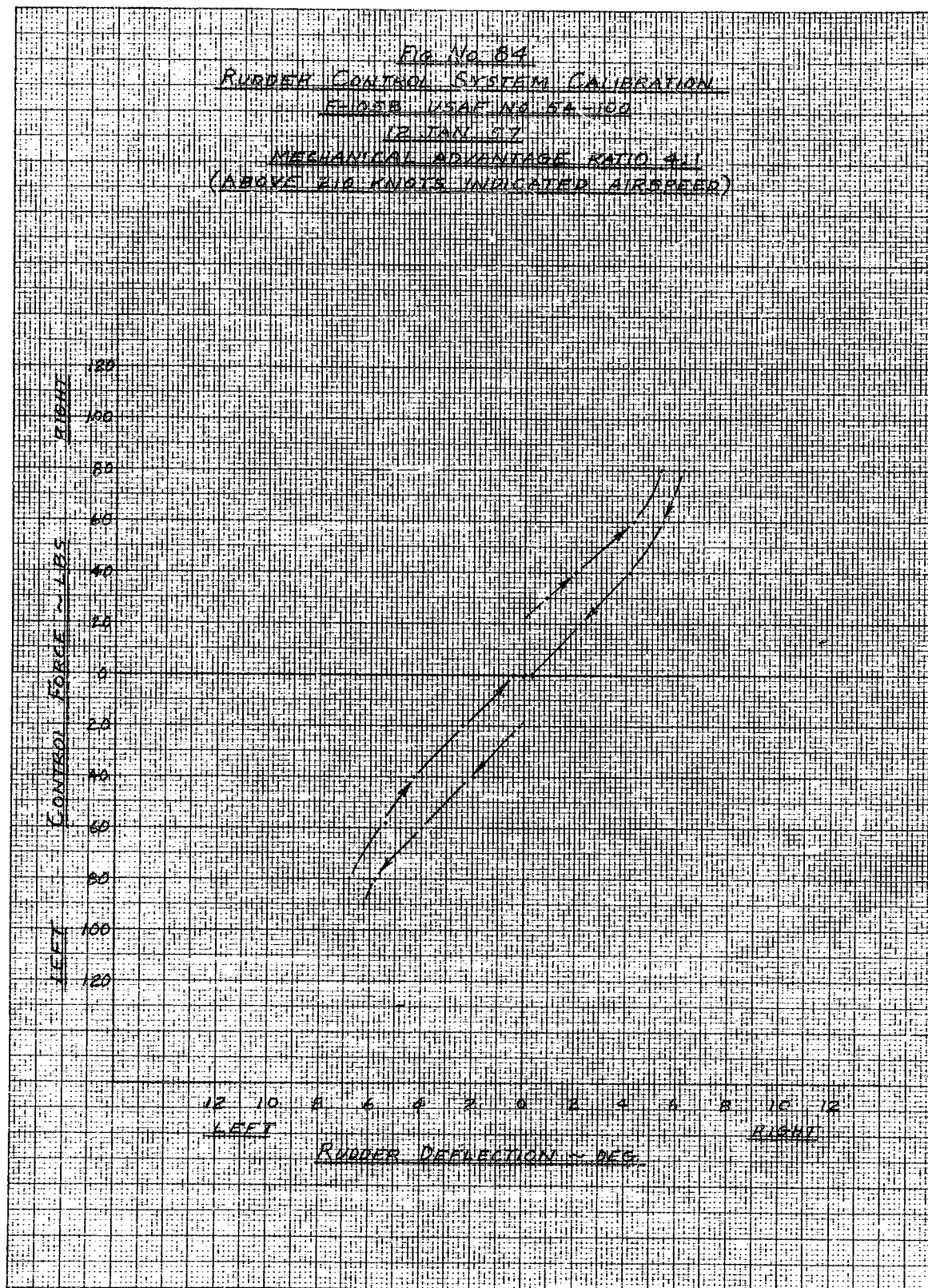


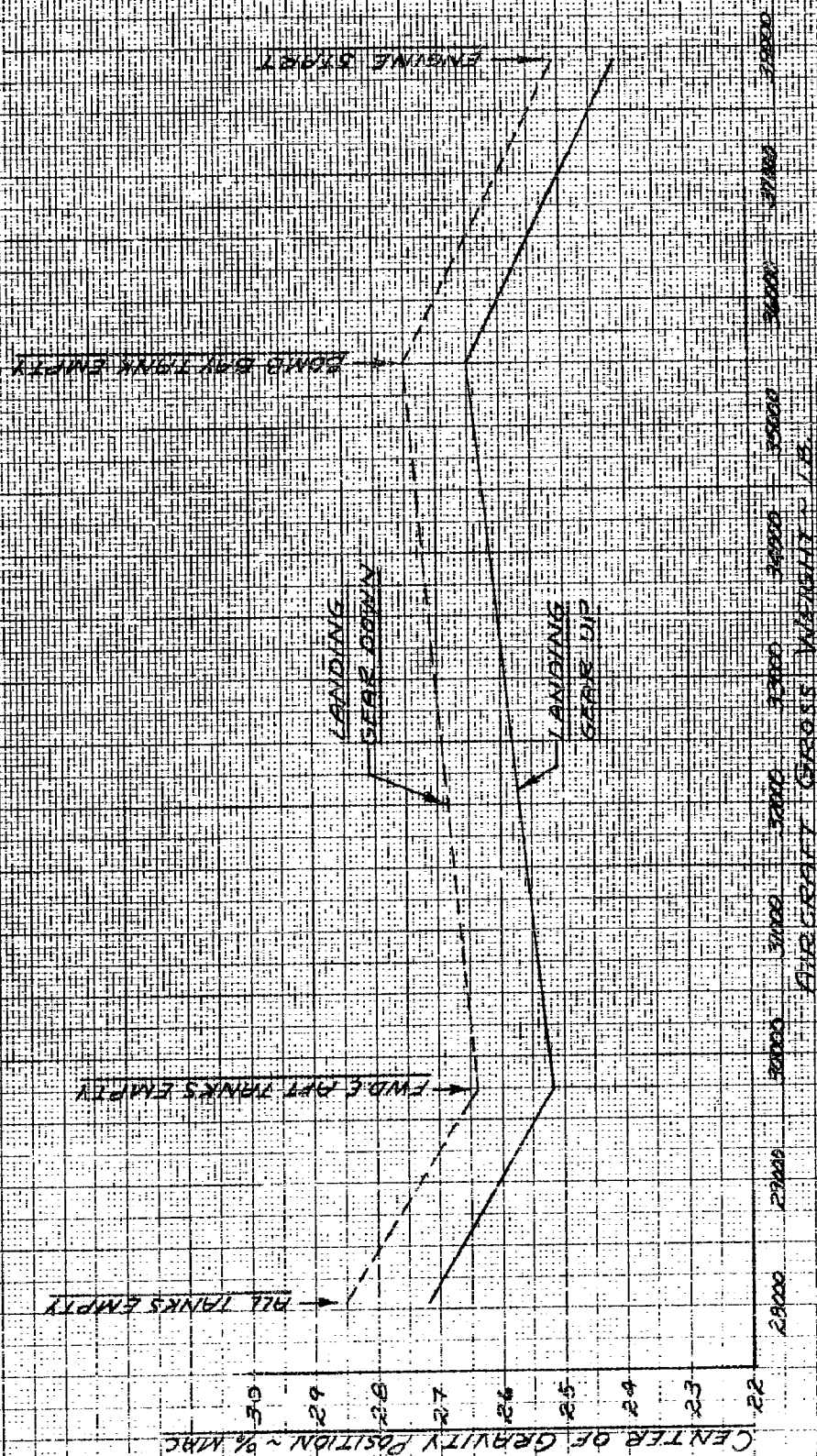
FIG. NO. 85

CENTER OF GRAVITY POSITION VS GROSS WEIGHT

F-105B WEIGHT 10,500

BOMB BAY TANKS INSTALLED

NO EXTERNAL STORES



APPENDIX II

general aircraft information

■ design information

From detail specification ES-347.

Overall

length 61 ft 1.328 in.
height over highest
part of tail 19 ft 8.38 in.

Wing

area 385.0 sq ft.
span 34 ft 11.2 in.
chord
at root 15 ft.
at station 210 7 ft.
MAC 11.46 ft.
airfoil section of
station 80 NACA 65A-005.5
airfoil section at
station 210 NACA 65A-003.7
incidence 0°
sweepback at 25
percent chord 45°
dihedral (negative) 3.5°
aspect ratio 3.18

Spoilers

span (normal
to airplane C_L) 109 in.
chord (percent
wing chord) 12%

Ailerons

span (normal to
airplane C_L) 58.58 in.
chord (parallel to
airplane C_L) (average
percent wing chord) 20%

Leading Edge Flaps

type plain
span normal to
airplane C_L (each) 117.63 in.

chord parallel to
airplane C_L (percent
chord) 11%

Speed Brakes

area approximately 28 sq ft
location aft fuselage

Maneuvering Stabilizer

span (normal to
airplane C_L) 17 ft 4.72 in.
chord (at airplane C_L) 7.5 ft.
airfoil section at root NACA 65A006
airfoil section at tip NACA 65A004
sweepback at
25 percent chord 45°
dihedral 0°
aspect ratio 3.05

Vertical Tail

airfoil section at root NACA 65A006
airfoil section at tip NACA 65A004
sweep of
25 percent chord line 43.5°
aspect ratio 1.786

Control and Control Surface Movements

rudder 32° right and left
rudder pedals 3.25 in forward and
aft
stabilizer 7° up and 25° down
7.17 in aft, 4.26 in
control stick forward
spoilers 61° up
spoiler stick
movement 7.17 in right and left
aileron movement 20° up and 20° down
wing leading edge
flaps 20° down
wing trailing edge
flaps 46° down
speed brake movement 50°

Fuel Tank Capacities — US Gals. (usable fuel)

tank	usable fuel
main	276.6
forward	369.4
aft	538.5
bomb bay	350.0
right inboard pylon	450
left inboard pylon	450
fuselage pylon	450

■ flight limits

Limit Diving Speeds	
5,000 feet	970 kts. TAS
25,000 feet	1100 kts. TAS
Airspeed, Gear Extended	243 kts. IAS
Limit Load Factors (33,070 lbs. gross weight)	
maneuver (subsonic)	+ 7.33, -3.0
maneuver (supersonic)	+ 7.33, -3.0
gust	+ 2.37, -0.37
Center of Gravity Position Limits	
forward	20.5% MAC
aft	35% MAC
Landing Limits (after aborted take-off)	
gross weight	37,964 lbs.
load factor at cg	2.2

■ engine limits

Maximum EGT (P and W Oper. Inst. 149)

Power	Operational Limits	Time Limits
Maximum	590	15
Military	590	30
Normal Rated	540	Continuous
90% Normal Rated	500	Continuous
80% Normal Rated	460	Continuous
70% Normal Rated	410	Continuous
Idle	340

■ weight and balance

The aircraft was weighed inside a closed hangar, in level attitude, with gear down, canopy closed, and with full fuel and oil and with test instrumentation and ballast installed. The gross weight at engine start was 37,880 pounds at a center of gravity position of 25.1 percent MAC. The landing gear up center of gravity position at this weight was 24.1 percent MAC.

Center of gravity position versus weight is shown in Figure 84, Appendix I.

■ test instrumentation

Test instrumentation was installed and calibrated by Republic Aircraft Corporation personnel and spot-checked by AFFTC personnel. The instrumentation was the same that had been installed by Republic Aircraft Corporation for the Phase I tests. A photo-panel and an oscillograph were installed in the nose compartment. A test airspeed boom was installed on the nose of the airplane and was equipped to measure angle of attack and angle of sideslip during flight. An AFFTC modified type D-1 pitot static head was used on the test nose boom. The pilot's fuel tank gages were used to obtain fuel transfer data to determine center of gravity positions in flight. Prior to the first flight of the test, two airspeed indicators, two altimeters, and an accelerometer were calibrated by AFFTC personnel and installed in the airplane for the Phase II program. The calibrated instrumentation installed was as follows:

OSCILLOGRAPH RECORDED DATA

Normal acceleration (at cg)	Outside air temperature
Lateral acceleration (at cg)	Compressor bleed valve light
Rudder position	Pilot's signal light
Left spoiler position	Oscillograph operate light
Right spoiler position	Intervalometer counter
Left aileron position	Time (clock)
Right aileron position	Time (stopwatch)
Stabilizer position	Normal acceleration
Angle of attack	P ₁ fin duct (upper)
Angle of sideslip	P ₁ fin duct (upper)
Rudder hinge moment	P ₁ fin duct (lower)
Stabilizer hinge moment	P ₁ fin duct (lower)
Lateral stick force	Bleed door position
Longitudinal stick force	Inlet duct plug position
Lateral stick position	P ₁ inside aft shroud (manifold)
Rate of roll	Correlation counter
Rate of pitch	Afterburner operation light
Rate of yaw	COCKPIT VISUAL INDICATORS
Fuel flow (engine)	Airspeed (nose boom)
Fuel flow (afterburner)	Altitude (nose boom)

PHOTO RECORDER DATA

Airspeed (nose boom)
Altitude (nose boom)
Airspeed (wing boom)
Altitude (wing boom)
Turbine discharge total pressure (4 gages)
High pressure rotor rpm
Exhaust gas temperature
Compressor inlet total pressure (manifold)
versus nose boom
pitot total pressure

Exhaust gas temperature
RPM (percent)
Normal acceleration
Angle of attack
Angle of sideslip
P ₁ /P ₂ ratio
Machmeter
Fuel quantity

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W. L. ALLEN, R. M. WHITE, CAPTAIN,
USAF. 112 PAGES. JUNE 1957.
(AFFTC-TR-57-9)

The performance of the F-105B is superior to any fighter-bomber presently in service; however, the aircraft has a large number of deficiencies. Although a maximum speed of 1.95 Mach number can be attained, poor acceleration characteristics provide little tactical utility from speeds above 1.8 Mach number. The specific range of the aircraft is approximately 20 percent less at 35,000 feet than estimated by the contractor. Both the longitudinal and lateral control systems are unsatisfactory. Structural failures which occurred during

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JUN 09 2000

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FROM: HQ AFMC/SCDP
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1. The following reports listed in the attached HQ AFMC/PAX Memo, 5 Jun 00, paragraph 1, were reviewed and cleared for public release in accordance with AFI 35-101, 1 Dec 99, *Public Affairs Policies and Procedures* (Case AFMC 00-104).

- AD016480 - *Rec'd 5/28*
- AD073570
- AD118718
- ADC051335

2. Please direct further questions to Lezora U. Nobles, HQ AFMC/SCDP, DSN 787-8583.

LEZORA U. NOBLES
AFMC STINFO Assistant
Directorate of Communications and Information

Attachment:
HQ AFMC/PAX Memo, 5 Jun 00, w/1 Atch